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THE ROYAL AERONAUTICAL SOCIETY

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THE ROYAL AERONAUTICAL SOCIETY

GRADUATE AND STUDENT SECTION

April 1950

LIFT SPOILERS FOR LATERAL CONTROL

by

R. R. DUDDY, B.Sc., A.F.R.Ae.S.

PART I

For certain types of aircraft, lateral control by means of lift spoilers offers several advantages compared with the orthodox aileron control. There are advantages for both very high speed aircraft with thin swept-back wings and for transport aircraft using moderately thick wing sections and high lift coefficients. At high speeds the serious reduction in control effectiveness, sometimes resulting in control reversal, due to wing twisting, can be avoided by spoilers because they cause much smaller changes in the local pitching moment than the normal aileron. For this reason spoilers are particularly attractive on aircraft using thin highly swept wings due to the difficulty of designing a high degree of torsional stiffness into wings of this sort. The chief advantage of spoilers on transport aircraft is that they allow very large span, or full span, flaps to be used. This is important because large span flaps are much more efficient than moderate span flaps in producing large increments in lift coefficient for small increases in drag coefficient. With spoiler controls the yawing moments which will accompany the rolling moments are more likely to be in the favourable direction than for normal ailerons. This advantage can be quite important when flying at high lift coefficients since, even if the adverse yawing moments associated with orthodox ailerons are not objectionable, they can reduce the rolling power seriously if the adverse yawing and resultant sideslip are not corrected by use of rudder.

In this short note the existing knowledge on lift spoilers, based on many wind tunnel tests backed up by some full-scale experience, is reviewed in order to examine the possibility of designing spoilers to give satisfactory lateral control.

Acknowledgment is made by the author to the Chief Scientist, Ministry of Supply, for permission to publish this note.

Requirements

Before considering the most important of the innumerable different types of lift spoiler which have been suggested, it is desirable to state the general requirements which a lateral control must satisfy.

(*i*) The maximum available rolling moment coefficient must be adequate; for a conventional transport aircraft this should be about 0.05. This requirement can generally be met quite easily with spoilers, due to the spoiler span being unrestricted by flap requirements. It is rather surprising that the rolling moment coefficient due to the spoiler with flaps retracted is roughly independent of the wing lift coefficient and continues to give a rolling moment of the desired sign even if the lift coefficient becomes negative. As speed increases the favourable yawing moment due to the spoiler becomes larger and this may become embarrassing at very high speeds.

(*ii*) The time lag between the start of the control movement and the start of the roll should be less than 0.2 sec. Ailerons always meet this requirement but lift spoilers will only do so at slow speeds if they are positioned within 0.3*c* of the wing trailing edge. This lag decreases as forward speed is increased and, in fact, it appears to be proportional to the time taken for a particle of air to travel from the spoiler to the wing trailing edge.

(*iii*) With some lift spoilers the time taken for the rate of roll to build up to its maximum value has been objectionably large. To avoid a criticism of "sluggishness" the control, in addition to having negligible lag, must be able to put on 20° of bank in less than 1.5 sec.

(*iv*) The rolling moment produced should increase steadily with increasing control deflection. Certain types of spoilers tend to be ineffective for small movements followed by a sudden increase in effectiveness at larger deflections. The tendency is most marked with flaps retracted at high speeds and at very low speeds with flaps down.

(*v*) The hinge moments also should increase steadily with increasing control deflection. Once again it is usually more difficult to satisfy this requirement with spoilers than with ailerons. The maximum control force required must be within the limits laid down for normal aileron controls.

There are many different types of lift spoilers but they can be divided into the following three groups:—

(*i*) Flat plate,

(*ii*) Circular arc, and

(*iii*) Slot lip.

Flat plate spoiler

As its name implies the flat plate spoiler consists of a rectangular plate with a hinge along one edge. This hinge is placed spanwise on the top surface of the wing, usually at a fixed percentage of the local chord from the leading edge. With the control neutral the plate lies flush with the wing surface and with full control movement the plate deflects to an angle of about 60°. The hinge moments for this type of spoiler are usually large (except for very small deflections) and require some form of balancing or power operation. With the hinge along the leading edge of the spoiler it may be over-balanced for small deflections, but with the hinge along the trailing edge it would be over-balanced at all deflections. An aircraft has flown successfully with a small unbalanced aileron connected to an over-balanced flat plate spoiler. A simple plate spoiler is usually ineffective for small movements but usually this can be overcome by having a duct from the bottom to the top surface of the wing arranged so that the spoiler seals the duct when the controls are neutral.

Circular arc spoiler

The circular arc spoiler was developed to eliminate the unpleasant hinge moments of the flat plate type. The hinge remains in the same position as before, but the flat plate is replaced by a circular arc concentric with the hinge. With the control neutral, the arc has to retract within the wing and this makes it slightly more difficult to install than the simple flat plate. Since the forces on the arc all pass through the hinge there is zero hinge moment, but the loads on the arc will be very large at high speed. It has been shown that satisfactory control "feel" can sometimes be provided artificially by springs, but with circular arc spoilers it is often difficult to prevent unpleasant "feel" due to high control inertia. As with the flat plate type this spoiler tends to be ineffective for small movements, unless there is a duct through the wing.

(*To be continued.*)

THE ROYAL AERONAUTICAL SOCIETY

4 HAMILTON PLACE, PICCADILLY, LONDON, W.1. GROSVENOR 3515-9

NOTICES APRIL 1950

ANNUAL GENERAL MEETING, 15th MAY 1950

Notice is hereby given that the Annual General Meeting of the Royal Aeronautical Society, with which is incorporated the Institution of Aeronautical Engineers, will be held on Monday, 15th May 1950, at 5.30 p.m., in the offices of the Society, 4 Hamilton Place, London, W.1.

AGENDA

1. To read the Notice convening the Meeting.
2. To receive and deliberate upon the Report of the Council on the state of the Society and the Balance Sheets and Income and Expenditure Accounts of The Royal Aeronautical Society and Aeronautical Trusts Limited for the year ended 31st December 1949.
3. To receive the names of those elected to Council for the years 1950-1953.
4. To announce the names of Fellows elected by the Council in accordance with Rule 4.
5. To elect the Auditors for the year 1950.
6. Any other business.

By Order of the Council,

J. LAURENCE PRITCHARD,

Secretary.

8*th April* 1950.

The President of the Society for the year 1950-51 takes office at the Annual General Meeting.

Note.—In accordance with the Rules any member whose subscription has not been paid before the first day of April is not entitled to vote.

Light refreshments will be served at 6 p.m.

PRESIDENT 1950-51

Major G. P. Bulman, C.B.E., B.Sc., *Fellow*, has been elected President of the Society for 1950-51.

CONTENTS OF APRIL JOURNAL

Design of Tail Pipes for Jet Engines—Including Reheat, J. L. Edwards, B.Sc., A.M.I.Mech.E.

Destructive Energy in Aircraft Pressure Cabins, P. B. Walker, M.A., Ph.D., F.R.Ae.S.

Electric Resistance Welding and its Use in the Construction of Aircraft, E. C. Sykes.

Analytical Treatment of Linked Levers and Allied Mechanisms, B. C. Carter, F.R.Ae.S.

Reviews. Correspondence.

The Council have set aside an annual sum of £250 for the award of premiums for papers published in the JOURNAL *and hope that members and non-members of the Society will contribute papers on their own special subjects.*

THE AERONAUTICAL QUARTERLY—Volume II

Volume II, Part I of "The Aeronautical Quarterly" will be published in May.

The contents will be:—

Two-Dimensional Theory of Stiffened Plates	Chang O'Chou
The Whirling of a Spinning Top	J. Morris
The Possibility of the Determination of Rate of Climb from Acceleration Measurements in Level Flight ...	E. C. Pike
A Note on Subsonic Aerofoil Theory	John W. Miles
Theory of an Oscillating Supersonic Aerofoil	Geoffrey L. Sewell
The Linearised Theory of Conical Fields in Supersonic Flow, with Applications to Plane Aerofoils	S. Goldstein and G. N. Ward

Copies of Part I, Volume II will be available from the offices of the Society towards the end of May at 7s. 9d. each to members of the Society, post paid, or 10s. 3d. each to non-members, post paid.

Volume I of "The Aeronautical Quarterly" (copies of which are still available) has been well received both in Great Britain and overseas; the response has been encouraging and reviews published both in this country and the United States have praised and welcomed both the appearance and contents of the Society's new publication. It is hoped that the circulation will steadily increase, thereby enabling the contents of each number to be still further improved and enlarged.

The special attention of members is drawn to the fact that a strictly limited edition only of "The Aeronautical Quarterly" is printed and numbers can not be reprinted in any circumstances. An annual subscription is therefore advisable.

GARDEN PARTY, 14th MAY 1950

The Society's Annual Garden Party will be held on Sunday, 14th May 1950, at White Waltham Aerodrome, near Maidenhead, from 2.30 p.m. to 6.30 p.m.

Application forms for tickets have been sent individually to members, who are reminded that the last day of application is the **10th May 1950.** The number of tickets is strictly limited and each member will be allowed four tickets only (10s. each, including tea). There will be a special train from Paddington and special buses from Maidenhead Station to the Aerodrome. Special cheap rail tickets will be obtainable **from Paddington only** and bus tickets **on the buses.**

The more leisurely flying days of the 1920s and 1930s—and before—of the flying programme will revive memories for many members and in addition there will be more modern aspects of flying and aeronautical activities and a remarkable display of valuable scale models showing the history of aircraft in Great Britain.

DATA SHEETS ON FUELS, LUBRICANTS AND HYDRAULIC FLUIDS

In addition to Data Sheets on Aerodynamics, Stressed Skin Structures and Performance, the Society is now producing a new series to provide information on the physical and chemical properties of aviation fuels, lubricants and hydraulic fluids, in a form convenient to aircraft and engine designers and users of aircraft and engines.

Data Sheets on Aviation Fuels will be ready in May 1950. They provide information on standard methods of test, a summary of the quality requirements of British and United States gasolines and distillate fuels and data on their composition, vapour, pressure, specific heat, viscosity, inflammability, and so on. The scope of the information required and its presentation in the form of Data Sheets have been the responsibility of the Fuels and Oils Sub-Committee of The Royal Aeronautical Society, on which the Institute of Petroleum has been represented. The Aviation Products Data Committee of the Institute has collected and analysed the Data.

The price of the Data Sheets on Aviation Fuels (seventeen in number), complete with folder, will be 23s. 6d. to members of the Society and of the Institute of

Petroleum, or 35s. 6d. to non-members. They will be obtainable in May by application, in writing, to the Secretary, The Royal Aeronautical Society, 4 Hamilton Place, London, W.1.

3rd LOUIS BLERIOT LECTURE

The Third Louis Bleriot Lecture will be read in Paris before the Association Française des Ingénieurs et Techniciens de l'Aéronautique (A.F.I.T.A.) on Thursday, 4th May 1950, by Sir Frederick Handley Page. The title of the Lecture will be "Economic and Safe Flight."

Any members of the Society who are in Paris will be welcomed at the Lecture by A.F.I.T.A.

38th WILBUR WRIGHT MEMORIAL LECTURE

The 38th Wilbur Wright Memorial Lecture will be read on Thursday, 25th May 1950, at 6 p.m., at the Institution of Civil Engineers, Great George Street, London, S.W.1, by Sir Richard Fairey, M.B.E., F.R.Ae.S. The title of the Lecture will be "Some Aspects of Expenditure on Aviation."

FELLOWS OF THE ROYAL SOCIETY

The following members of the Society have recently been made Fellows of the Royal Society:—

R. A. Fraser, B.A., D.Sc., *Fellow*; S. B. Gates, O.B.E., *Fellow*, and L. Howarth, B.Sc., M.A., Ph.D., *Associate Fellow*.

MEMBERS' NEW APPOINTMENTS

K. T. Spencer, B.Sc., A.C.G.I., *Fellow*—Deputy to the Chief Scientist at the Ministry of Fuel and Power.

H. L. Stevens, *Fellow*—Director of Military Aircraft Research and Development, Ministry of Supply.

J. E. Serby, *Fellow*—Deputy Director at the Royal Aircraft Establishment.

A. W. Rogers, *Associate Fellow*—Experimental Officer in Technical Development Plans Department, Ministry of Supply.

J. R. Neill, *Associate*—Commandant at Renfrew Airport.

ASSOCIATE FELLOWSHIP EXAMINATIONS

The next Associate Fellowship Examination in the New Syllabus will be held on 31st May, 1st and 2nd June 1950. All candidates who have entered for the Examination will be informed individually of the detailed arrangements.

Those candidates who have been given another opportunity to complete the examination under the Old Syllabus will have full particulars of the examination to be held at the end of August.

ELLIOTT MEMORIAL PRIZE

The Elliott Memorial Prize has been awarded to F. II.A. Corporal 583943 Armitage, M. J., the aircraft apprentice of the May 1947 Entry who has obtained the highest marks in the General Studies Examination.

LECTURES—SPRING 1950

MAIN LECTURES

(At 6 p.m. at the Institution of Civil Engineers, Great George Street, S.W.1, unless otherwise stated. Tea will be served at 5.30 p.m.)

Thursday, 13th April 1950—The Berlin Air Lift, Air Commodore J. W. F. Merer, C.B.

Thursday, 27th April 1950—AT MANCHESTER—Recent Developments in High Speed Research in the Aerodynamics Division of the N.P.L., J. A. Beavan, M.A.(Cantab), A.F.R.Ae.S., and D. W. Holder, D.I.C., A.C.G.I., B.Sc, A.F.R.Ae.S.—**at the Chemistry Lecture Theatre, Manchester University, at 6.30 p.m.**

Thursday, 4th May 1950—3rd LOUIS BLERIOT LECTURE—IN PARIS, Sir Frederick Handley Page, C.B.E., F.I.Ae.S., Hon.F.R.Ae.S.

Thursday, 25th May 1950—38th WILBUR WRIGHT MEMORIAL LECTURE, Some Aspects of Expenditure on Aviation, Sir Richard Fairey, M.B.E., F.R.Ae.S.

SECTION LECTURE

(*At 7 p.m. in the Library, 4 Hamilton Place, London, W.1.*)

Tuesday, 25th April 1950—Flight Refuelling and the Problem of Range, C. H. Latimer Needham, M.Sc., F.R.Ae.S.

GRADUATE AND STUDENT SECTION

Tuesday, 18th April 1950—A Discussion Evening on The Training of Aeronautical Engineers. Chairman: Marcus Langley, F.R.Ae.S. *All members of the Society are particularly invited to take part in this discussion.*
In the Library, 4 Hamilton Place, at 7.30 p.m.

BRANCH LECTURES

BIRMINGHAM BRANCH

Friday, 21st April 1950—Annual General Meeting and Smoking Concert—*at the White Horse Hotel.*

BRISTOL BRANCH

Monday, 24th April 1950—Annual General Meeting and Film Show.
In the Conference Room, Bristol Aeroplane Co. Ltd., Filton House, at 6 p.m.

COVENTRY BRANCH

Wednesday, 19th April 1950—Annual General Meeting and Films.
In the Trinity Hall, Ford Street (near Pool Meadow) at 7.30 p.m. Membership cards must be shown at all lectures.

GLASGOW BRANCH

Tuesday, 18th April 1950—Structural Testing, J. B. B. Owen, B.Sc., A.F.R.Ae.S.
At the Royal Technical College, Glasgow, at 7.30 p.m.

GLOUCESTER AND CHELTENHAM BRANCH

Thursday, 20th April 1950—*at Cheltenham*—Annual General Meeting.
In the Empire Hall, North Street, at 7.30 p.m.

LEICESTER BRANCH

Wednesday, 19th April 1950—Short papers by members.
In Room 104, Leicester College of Technology, at 7.15 p.m.

PORTSMOUTH BRANCH

Friday, 21st April 1950—The Probable Role and Influence of Aircraft in Future Warfare, Air Marshal Sir Robert Saundby, K.B.E., C.B., M.C., D.F.C., A.F.C.
In the Lecture Room, Central Library, at 7 p.m.

PRESTON BRANCH

Wednesday, 26th April 1950—Annual General Meeting.
In the Assembly Hall of the Technical College, Corporation Street, at 7.30 p.m.

READING AND DISTRICT BRANCH

Wednesday, 12th April 1950—Flying Boats, H. Knowler, F.R.Ae.S., A.M.I.C.E.
At the Abbey Gateway Rooms at 7.30 p.m.

WEYBRIDGE BRANCH

Wednesday, 31st May 1950—Annual General Meeting.
At Vickers-Armstrongs Ltd., Weybridge Works, at 6 p.m., unless previous notice is given.

LAPEL BADGES

Lapel badges are still available to members at 3s. 6d. each, including postage. These badges are gilt enamel with a screw back and are $\frac{3}{4}$ inch in diameter. Remittances should be sent, with applications, to the Secretary at the Offices of the Society.

JOURNAL BINDING

The new prices of binding of Journals is as follows:—

1949 Volume (including packing and postage), 15s. 6d.

Previous Volumes (including packing and postage), 17s. 0d.

Journals should be sent direct to the Lewes Press, Friars Walk, Lewes, Sussex, and the remittance to the Secretary at the offices of the Society.

Cases only for unbound 1949 Volumes are available, price 6s. 6d. each. Requests for cases, with remittances, should be sent to the Secretary at the offices of the Society.

INCOME TAX

In response to numerous inquiries from members with regard to a rebate on Income Tax for their subscriptions, the following is a copy of a letter received from the Principal Inspector of Taxes.

Ref. H.R.S. 34/C.I. 4420/63.

Dear Sir,

Further to your interview with Mr. Stonely at this office on the 19th July, I am now in a position to inform you that the Board of Inland Revenue will not raise objection to the allowance as an expense for Income Tax purposes of annual subscriptions paid by members who are:—

(*i*) Assessable under Schedule D of the Income Tax Acts in respect of professional or trading profits, subject to the decision of the Commissioners who make the assessment that such subscriptions are sufficiently closely related to the business on: or

(*ii*) Assessable under Schedule E in those cases only in which continual membership of the Society is an essential condition of the terms of appointment.

Yours faithfully,

(Signed) GEO. WILCOCK,

Principal Inspector of Taxes.

CHANGES OF ADDRESS

To assist in keeping the records of members correct and up to date the Secretary will be glad if all members will notify him as soon as possible of changes of address.

When notifying changes please give the following particulars:—

Name (in block letters).

Grade of membership.

New address (in block letters).

Old address.

Changes of address must be received *before the 15th of the month* in order to be effective for the Journal for the following month.

ELECTIONS

The following is a list of new members and transfers of membership of the Society:—

Associate Fellows

Charles Frederick Nellthorpe, Robert Emmanuel Moger, Herbert Henry Wakinson (from Associate).

Associates

Woulfe Hay Hicks, Frederick Arthur Jolly, Clarence Hamilton McVeigh, William Frederick Stevens.

Graduates

Beryl Mary Bigg, Peter George Bullen (from Student), Peter Field, John Dennis Leyland Gregory, Brian Aubrey Harding, Hubert William Henry Harding, Peter John Harvey (from Student), Francis Farrell Heaton (from Student), Guy Thornton Smith (from Student), Ronald John Stephens, Edward Philip Vaughan, Brian Hawley Wager (from Student).

Companion

John Cranfurd Powell.

ACKNOWLEDGMENTS

The Council acknowledge with grateful thanks the gift of books from H. R. Trost, Esq., and R. J. Lawes, Esq., *Associate*; and the return of back numbers of the JOURNAL from Miss A. D. Betts, *Companion*; Major G. Dennison, *Founder Member*; and W. R. Hudson, Esq. (*ex-Student*).

ADDITIONS TO THE LIBRARY

The following have been added to the Library (titles of Pamphlets are printed in italics and books marked * or ** may not be taken out on loan):

Method of Presentation of Data for Proposed Aircraft. Air Transport Association. A.T.A. 1949.

The Complete Air Navigator. D. C. T. Bennett. 5th Edition. Pitmans. 1950.

Aeronautics—Heavier-Than-Air Aircraft. Part I. M. J. B. Davy. H.M.S.O. 1949.

Photoelasticity—Volume I. M. M. Frocht. Wiley and Sons. 1941.

Aerials for Centimetre Wave-Lengths. D. W. Fry and F. K. Goward. Cambridge University Press. 1950.

Technical Sketching and Visualization for Engineers. H. H. Katz. Macmillan, New York. 1949.

The Conquest of Space. W. Ley and C. Bonestell. Viking Press. 1950.

Nationalisation in Practice. The Civil Aviation Experiment. J. Longhurst. Temple Press. 1950.

Airplane Performance, Stability and Control. C. D. Perkins and R. E. Hage. Wiley and Sons. 1949.

Noise and Sound Transmission. Physical Society. Physical Society. 1949.

Equilibrium Data for Tin Alloys. Tin Research Institute. T.R.I. 1950.

Aerial Mapping and Photogrammetry. L. G. Trorey. Cambridge University Press. 1950.

Theory and Design of Gas Turbines and Jet Engines. E. T. Vincent. McGraw-Hill. 1950.

Plasticity in Engineering. F. K. Th. van Iterson. Blackie and Son. 1947.

Sound Absorbing Materials. C. Zwikker and C. W. Kosten. Elsevier Publishing Co. 1949.

N.A.C.A. Technical Notes

1949—*Annular-jet ejectors. E. G. Reid.*

1966—*Experimental investigation of flutter of a propeller with Clark Y section operating at zero forward velocity at positive and negative blade-angle settings. J. E. Baker and R. S. Paulnock.*

1967—*A comparison of wing loads measured in flight on a fighter-type airplane by strain-gage and pressure distribution methods. W. S. Aiken, Jnr., and D. A. Howard.*

1979—*A radar method of calibrating airspeed installations on airplanes in maneuvers at high altitudes and at transonic and supersonic speeds. J. A. Zalovcik.*

1981—*Critical stress of ring-stiffened cylinders in torsion. M. Stein, J. L. Sanders, Jnr., and H. Crate.*
1982—*An introduction to the physical aspects of helicopter stability. A. Gessow and K. B. Amer.*
1983—*Longitudinal flying qualities of several single-rotor helicopters in forward flight. F. B. Gustafson, K. B. Amer, C. R. Haig and J. P. Reeder*
1984—*A semiempirical method for estimating the rolling moment due to yawing of airplanes. J. P. Campbell and A. Goodman.*
1986—*Stability of alclad plates. K. P. Buchert.*
1987—*An analysis of supersonic aerodynamic heating with continuous fluid injection. E. B. Klunker and H. R. Ivey.*
1988—*Transient behaviour of lumped-constant systems for sensing gas pressures. G. J. Delio, G. V. Schwent and R. S. Cesaro.*
1989—*Some effects of density and Mach number on the flutter speed of two uniform wings. G. E. Castile and R. W. Herr.*
1990—*Plastic buckling of a long flat plate under combined shear and longitudinal compression. E. Z. Stowell.*
1991—*Theoretical loading at supersonic speeds of flat swept-back wings with interacting trailing and leading edges. Doris Cohen.*
1992—*Linearized lifting-surface theory for swept-back wings with slender plan forms. H. Lomax and M. A. Heaslet.*
1993—*Icing-protection requirements for reciprocating-engine induction systems. W. D. Coles, V. G. Rollin and D. R. Mulholland.*
1994—*Knock-limited performance of fuel blends containing aromatics. VI-10 alkylbenzenes. I. L. Drell and H. E. Alquist.*
1996—*On internal damping of rotating beams. M. Morduchow.*
1997—*Analysis of means of improving the uncontrolled lateral motions of personal airplanes. M. O. McKinney, Jnr.*
1998—*Aerodynamic characteristics of the NACA 8-H-12 airfoil section at six Reynolds numbers from* 1.8×10^6 *to* 11.0×10^6. *R. F. Schaefer and H. A. Smith.*
2009—*Theoretical calculations of the supersonic pressure distribution and wave drag for a limited family of tapered sweptback wings with symmetrical parabolic-arc sections at zero lift. J. H. Kainer.*

N.A.C.A. Technical Memoranda

1220—*Pressure-distribution measurements on the tail surfaces of a rotating model of the design BFW-M. 31. M. Kohler and W. Mautz.*
1237—*On the motions of an oscillating system under the influence of flip-flop controls. I. Flugge-Lotz and K. Klotter.*
1243—*Two-dimensional potential flows. M. Schafer and W. Tollmien.*
1244—*Rotationally symmetric potential flows. M. Schafer and W. Tollmien.*
1247—*Landing procedure in model ditching tests of Bf.10. W. Sottorf.*
1254—*Systematic model researches on the stability limits of the DVL series of float designs. W. Sottorf.*

N.A.C.A. Reports

867—*A theoretical investigation of hydrodynamic impact loads on scalloped-bottom seaplanes and comparisons with experiment. B. Milwitzky.*
879—*Interference method for obtaining the potential flow past an arbitrary cascade of airfoils. S. Katzoff, R. S. Finn and J. C. Laurence.*
893—*Velocity distributions on two-dimensional wing-duct inlets by conformal mapping. W. Perl and H. E. Moses.*
895—*Analysis of variation of piston temperature with piston dimensions and undercrown cooling. J. C. Saunders and W. B. Schramm.*
896—*Pressure-sensitive system for gas-temperature control. R. S. Cesaro and N. Matz.*

900—*The use of source-sink and doublet distributions extended to the solution of boundary-value problems in supersonic flow. M. A. Heaslet and H. Lomax.*

A.R.C. Reports and Memoranda

2289—*Axial or longitudinal vibration of an aircraft. J. Morris, W. A. Jones and G. F. Walton.*

2299—*The vulnerability of metal aircraft structures. R. W. Brown.*

2316—*An application of strain gauges to the measurement in high-speed wind tunnels. P. J. Wingham.*

2323—*Experiments on tail flutter. C. Scruton.*

2372—*Flight tests to investigate the lightness of an unbalanced elevator on the Lockheed 12A. M. B. Morgan and D. E. Morris.*

2430—*The stresses in a flat panel under shear when the buckling load has been exceeded. D. M. A. Leggett.*

College of Aeronautics

32—*An empirical method for rapidly determining the loading distributions on swept back wings. R. Stanton Jones.*

34—*The characteristics of systems which are nearly in a state of neutral stability. W. J. Duncan.*

Nationaal Luchtvaartlaboratorium, Amsterdam

S.347—*Load distribution and relative stiffness parameters for a reinforced flat plate containing a rectangular cut-out under plane loading. L. S. D. Morley.*

Publications Scientifiques et Techniques du Ministere de l'Air

234—*L'étude des jets et la mécanique théorique des fluides. A. Oudart.*

235—*Nouvelles méthodes de préparation des cristaux uniques d'aluminium et leurs applications a l'etude microscopique et cristallographique de l'état métallique. L. Beaujard.*

B.S.T.112—*Détermination d'un état plan des contraintes a l'aide d'un extensiomètre a résistance élestrique a trois directions. Abaques pratiques d'emploi. F. T. Salles.*

J. LAURENCE PRITCHARD,
Secretary.

THE LEWES PRESS (Wightman & Co. Ltd.), Friars Walk, Lewes, Sussex.

SILICONES

are already important to the

AIRCRAFT INDUSTRY

Silicones have heat-resisting and water-repellent properties that have important applications in the Aircraft Industry. They can be used in the form of Liquids, Resins, Greases and Rubbers (Silastic).

SILICONE FLUIDS are subject to low viscosity changes over wide temperature ranges. They are suitable for damping, hydraulic and instrument fluids.

SILICONE RESINS AND VARNISHES are heat-stable and water-repellent. They provide electrical insulation that is serviceable at temperatures at least 50°C. above Class B limits.

SILICONE GREASES AND COMPOUNDS are lubricants suitable for bearings and valves operating at high and low temperatures. DC4, the Ignition Sealing Compound, has many applications.

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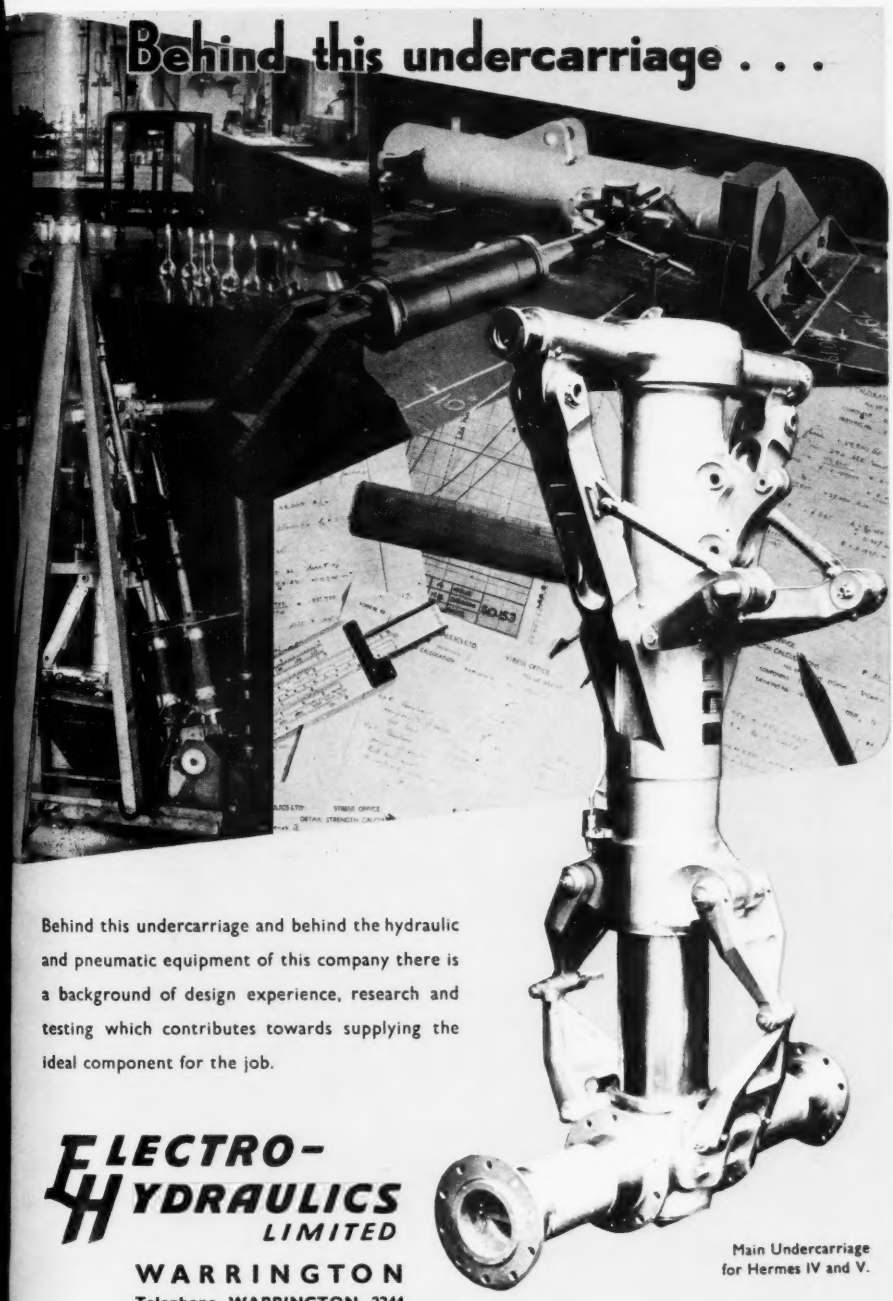

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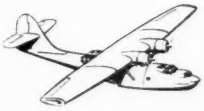

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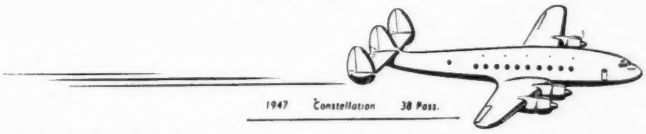

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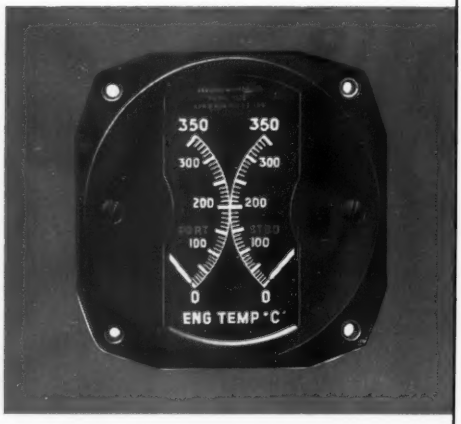

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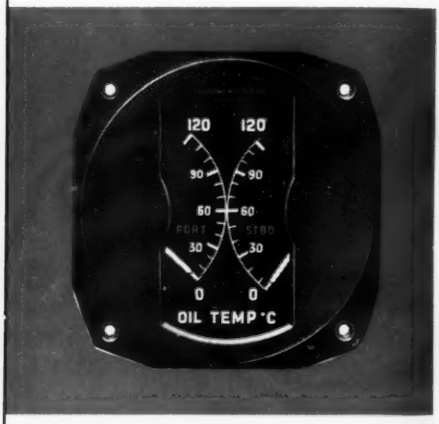

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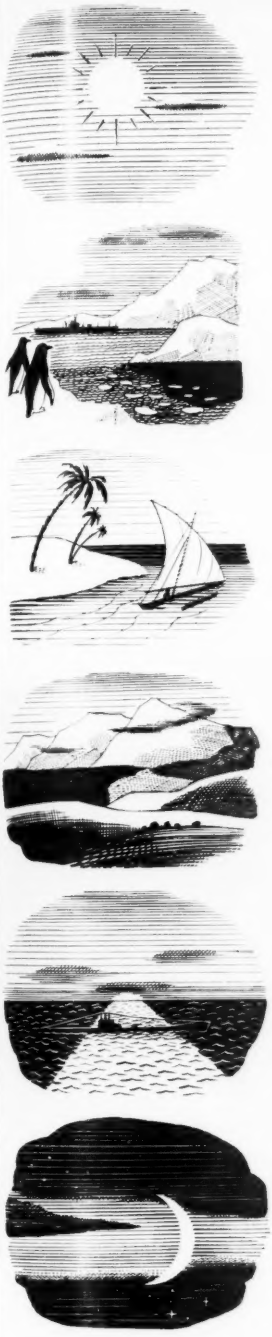

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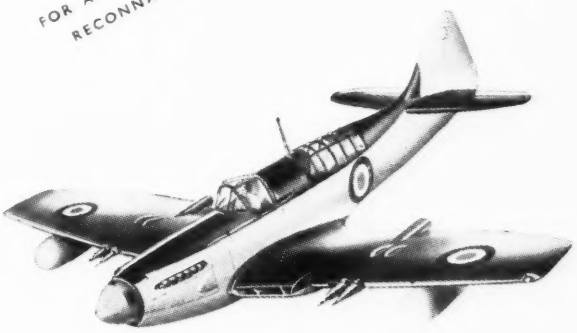

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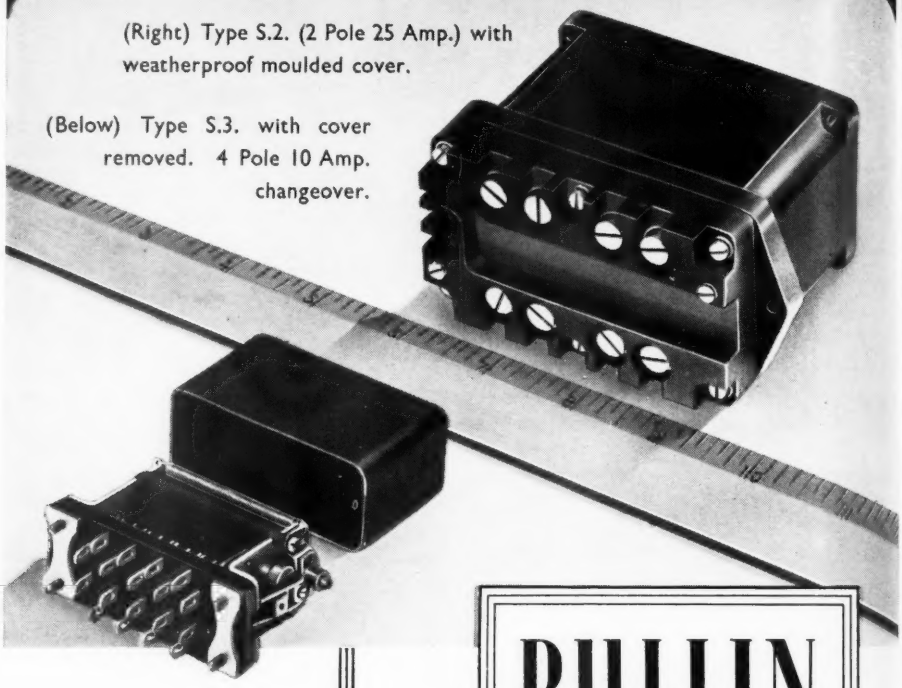

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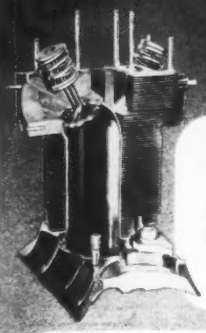

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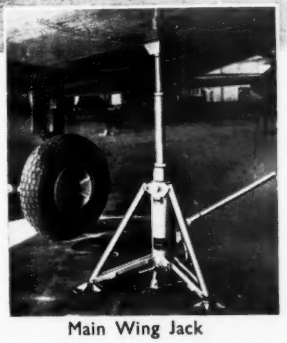

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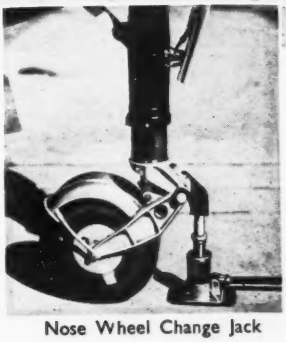

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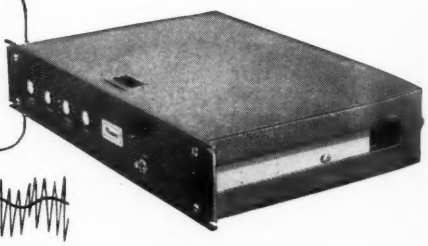

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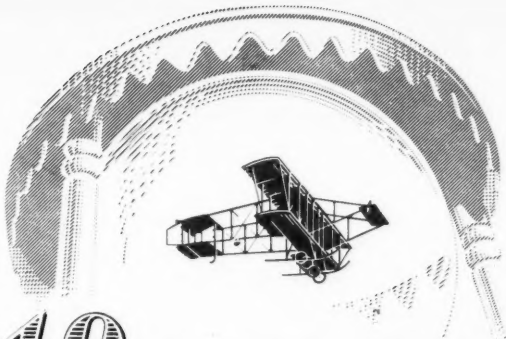

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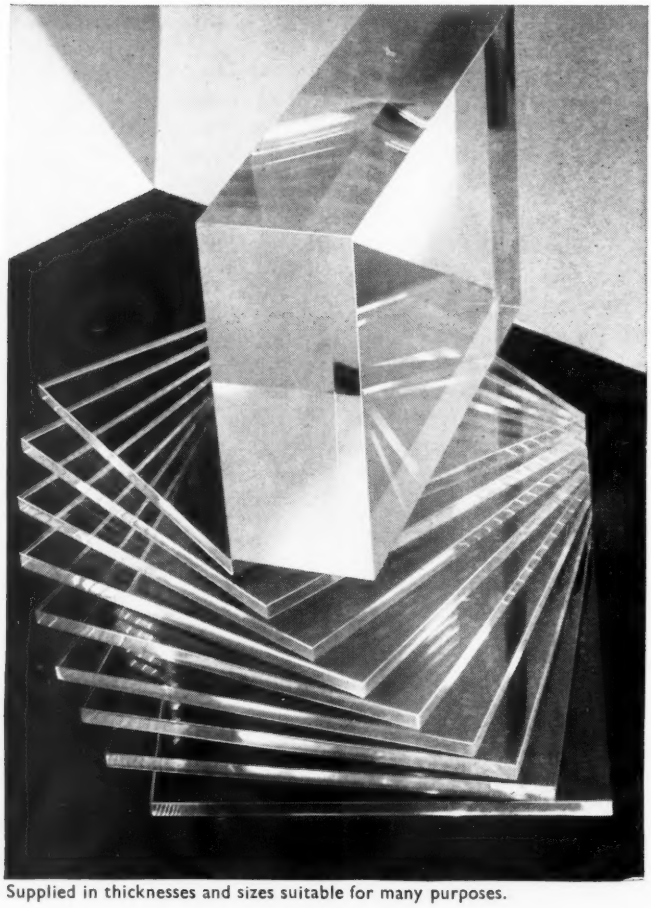

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The principle of the ram-jet or athodyd has been understood for a number of years. It has been applied in many ways, the most prominent being the flying bomb and, rather less spectacularly, the ducted radiator. The principle is to convert the velocity energy of a high speed air stream to pressure energy by means of a suitable duct, to apply heat energy to the compressed air, and then to expand this heated air in the form of a high velocity jet. The advantage of such a mechanism is an increase in thrust given by the product of the mass flow of air, and the difference in velocities between entry and exit; the disadvantages are, the necessity to project the duct at high forward speed before any thrust is obtained and, except at very high speeds indeed, the low thermal efficiency.

A possible application of the propulsive duct is to append it to the exhaust of a gas turbine engine and, by adding more heat to the gas stream, to eject it finally to atmosphere at an increased velocity, thus increasing the change of momentum over the engine/duct combination and hence, the total thrust. This is the principle of "reheat" or "after burning," and Fig. 1 illustrates diagrammatically its application to a typical jet engine.

In considering any increase in the performance of a jet engine, there are many factors to be borne in mind, not the least of which is the effect of such increases on the operating temperature throughout the engine cycle. If an alteration is made to the exhaust system of a normal turbo-jet having the effect of increasing the temperature of the gases passing through the propulsive nozzle, then, for a given size of nozzle, the mass flow will fall and the pressure in the exhaust cone will rise. These changes will be transmitted through the entire engine, and will result in the compressor moving nearer the surge point, and the pressure drop across the turbine decreasing. If the power output of the turbine is to be maintained under these changed conditions, a severe increase in turbine entry temperature will be necessary, with obvious mechanical complications.

If, concurrently with the increase of temperature at the final jet, there is an increase in the area of the propelling nozzle, the rate of mass flow and the turbine pressure drop may be left unchanged. This means that the compressor/turbine combination is unaffected by the application of reheat, but it also means that unless some means of reducing the nozzle size for normal running is incorporated, then a severe loss in thrust can be anticipated when the engine is operated without thrust boost. The obvious answer to this problem is a variable nozzle, an answer far from simple to realise reliably in practice.

A second important consideration, which will be dealt with later, is the total pressure loss incurred in the reheat combustion chamber. This pressure loss is caused by two factors: first, the skin friction losses due to flame stabilisers, additional length of tail pipe and higher scrubbing velocities, and secondly, the momentum loss inherent in any system where heat is supplied to a moving gas stream. The general effect of these losses is to decrease the total pressure at entry to the propulsive nozzle. From considerations of the flow through ducts, it can be shown that the Mach number at the end of the jet pipe will tend to increase, and in the extreme —if, for example, frictional losses were abnormally high—the Mach number at the end of the combustion zone would be unity and no propulsive nozzle would be necessary, or for that matter possible, if a pressure rise at the turbine is to be avoided.

* A Section Lecture given on 6th December 1949. This was the 788th Lecture to be read before the Society.

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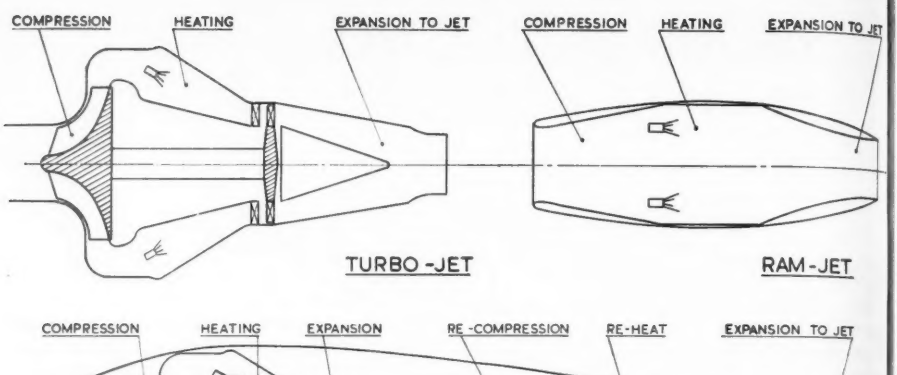

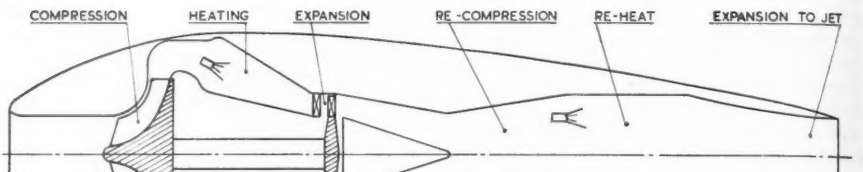

Fig. 1.
Sketch of turbo-ram-jet.

It can be demonstrated that economical operation will only be obtained if the velocity at entry to the reheat section is reduced to a minimum, and it will be shown later that any attempt to supply heat to a gas stream initially moving at a high Mach number is foredoomed to failure. The result is a high momentum pressure loss, a small thrust increase—or even in an extreme case, a thrust decrease—and thoroughly uneconomical operation.

It is possibly advisable at this point to review the problem briefly from a thermodynamic, rather than from a gas flow aspect.

The thermodynamic efficiency of any cycle depends on the temperature ratio, or alternatively the pressure ratio, of the operating points. Consider a fixed heat supply and a fixed exhaust pressure; the higher the pressure of the gases before heat is supplied, the greater will be the pressure ratio through which expansion subsequently takes place, and the greater will be the thermal efficiency. This optimum pressure for subsonic flow is limited to that given by the critical pressure ratio, and can only be obtained by diffusing the gases entering the reheat zone, to the lowest possible Mach number—and highest possible pressure—before adding heat. If this is done, the thermal efficiency of the reheat process will be a maximum and the added fuel consumption will be a minimum. This conclusion, in general, is the same as that reached from flow considerations.

When considering efficiency, the thermal and combustion efficiencies must not be confused. A high value of the thermal efficiency means an efficient cycle, and a high thrust for a given consumption of available oxygen; a high value of combustion efficiency means an economical fuel consumption, which in this application is of secondary importance when compared with the effect produced.

2. THEORETICAL SURVEY

2.1. Description of cycle

It is assumed that the equations of gas flow through ducts are appreciated, and no proofs are given. In general, the quantities of non-dimensional mass flow, velocity, total to static pressure and temperature ratios, and Mach number will be taken from families of curves similar to those in Ref. 1.

The usual symbols are employed as follows:—

A = area

P, T = total pressures and temperatures

p, t = static pressures and temperatures
V = velocity
W_a = air mass flow
W_f = fuel mass flow
γ = ratio of specific heats
$(')$ = conditions with reheat
M = Mach No.
D = diameter
L = length

Figure 2 shows diagrammatically a reheat pipe and the stations selected. Fig. 3 shows the cycle on a temperature-entropy diagram. The air enters the compressor at atmospheric pressure—possibly with pre-compression due to ram—and is compressed with some increase of entropy. Heat is then applied at nominally constant pressure, although there are bound to be some losses in the combustion system, and the gases are expanded through the turbine to point (1). The turbine outlet pressure and temperature have been selected as datum points in the reheat calculations and all figures quoted in the ensuing sections presuppose that this static point is unchanged.

With the normal engine the gases would now be expanded through the propelling nozzle to form the propulsive jet. With reheat the gases after diffusion to (3) are further heated to the temperature of point (4), but at the expense of some loss in total pressure, and are then expanded through the propelling nozzle to (5). As can be seen, the increase in temperature makes more energy available for conversion to kinetic energy, but it also increases the amount of heat rejected in the wake, with a resulting

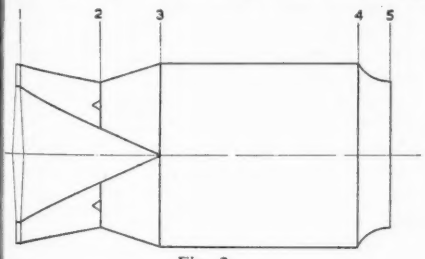

Fig. 2.
Stations for calculation and symbols.

AD	Area and diameter
V	Velocity
PT	Total pressure and temperature
pt	Static pressure and temperature
γ	Ratio of specific heats
M	Mach number
W_a	Mass flow of air (lb/sec)
W_f	Mass flow of fuel (lb/sec)
F	Thrust (lb)

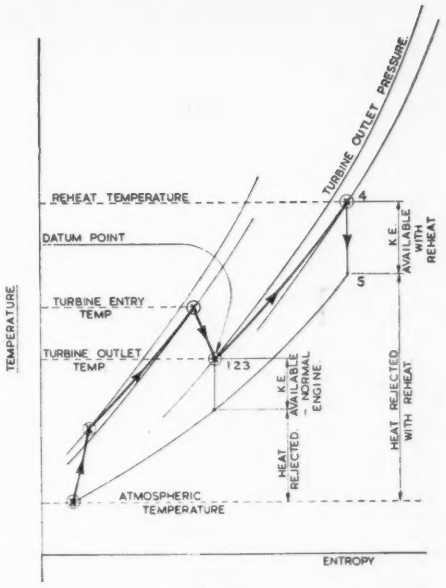

Fig. 3.
Temperature entropy diagrams of turbo-ram-jet.

increase in thrust at the expense of a decrease in thermal efficiency.

This gives a broad picture of the process, and in the following sections, symbols indicated with a prime (′) will denote conditions with reheat in operation. As a practical case, and one on which some test and flight figures are available, the Goblin engine in the Vampire has been selected for detailed consideration.

2.2. Simple Theory, Neglecting Losses

If all losses are neglected and it is stipulated that the Mach number in the propelling nozzle shall be less than unity and the same with and without reheat, turbine conditions being maintained steady by a variable nozzle, a simple relationship between proportionate thrust increase and nozzle size is obtainable. For a given Mach number the non-dimensional mass flow remains constant, and we can therefore write:

$$\frac{(W_a+W_f)\sqrt{T_5}}{A_5\, p_a} = \frac{(W_a'+W_f')\sqrt{T_5'}}{A_5'\, p_a}$$

From the assumptions made, and neglecting the fuel flow into the tail pipe:

$$A_5'/A_5 = \sqrt{(T_5'/T_5)}$$

In the same way, the non-dimensional velocities with and without reheat, are equal,

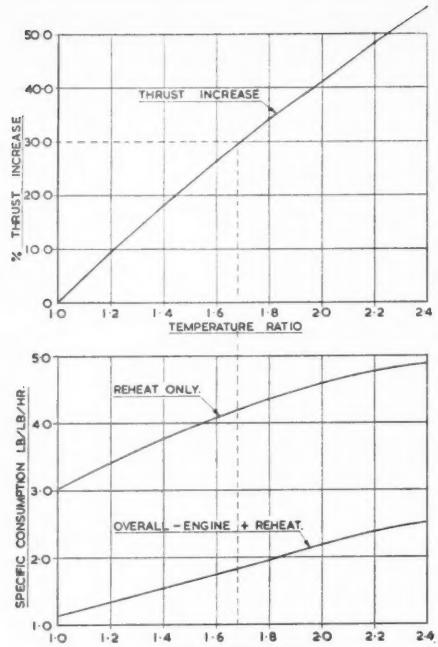

Fig. 4.
Theoretical performance of tail pipe with no losses.

whence:

$$V_5/\sqrt{T_5} = V_5'/\sqrt{T_5'}$$
$$V_5'/V_5 = \sqrt{(T_5'/T_5)}$$

The thrust for a given mass flow under static conditions is directly proportional to the jet velocity, which gives:

$$F'/F = V_5'/V_5 = \sqrt{(T_5'/T_5)}$$

or the thrust and nozzle size increase directly as the square root of the absolute gas temperature. The fuel consumption is given by the expression:

$$\left.\begin{aligned} W_f \times \text{Calorific Value} \\ = (W_a + W_f) \times C_p \times (T_5' - T_5) \end{aligned}\right\} \quad (4)$$

As might be expected, these figures differ somewhat from those obtained in practice, but for the sake of subsequent comparisons, values are shown on Fig. 4 as they would apply to a Goblin engine.

2.3. Fundamental Requirements of a Reheat System

It is a fundamental disadvantage of reheat as applied to any existing jet engine, that the specific fuel consumption will be high. The curves on Fig. 4 give minimum values for a particular application, and it will be seen that for an increase of 30 per cent. in S.L.S. (Sea Level Static) thrust, a reheat specific consumption of 4.20 lb./hr./lb. would be required. At the same time the increase in thrust will reduce the time to climb to a given altitude, thus tending to minimise the differences in amounts of fuel used.

The theoretical considerations so far made have assumed perfect conditions. In practice, and quite apart from "parasitic" losses due to friction and poor combustion, there is a fundamental loss which is incurred whenever heat is supplied to a moving gas stream. If a realistic design of tail pipe is to be produced, all factors must be considered, and a detailed analysis is given in the following sections.

These considerations will determine the dimensions and the anticipated performance of the system from an aerodynamic aspect. The design so produced will not be final, however, since modifications may be necessary if combustion is to be effective and efficient.

2.4. Fundamental Pressure Loss due to Heating

The initial design requirement of a reheat system will undoubtedly be the proportionate thrust increase, and as a particular example it is proposed to select 30 per cent., this being applied to the Goblin engine for which full performance data is available.

Figure 4 shows that for this case an ideal temperature ratio of 1.68 will be required. The actual temperature will be higher when all the necessary adjustments for the various losses have been made, but this quick estimate serves as a guide.

Assume now that the area of the combustion zone is constant and that the gases are evenly mixed as regards temperature and velocity at entry to and exit from this zone, and that all heat is supplied before the gases pass station (4).

Neglecting friction losses we may write from considerations of momentum, continuity and state:

$$\frac{W V_3}{g} + p_3 A_3 = \frac{W V_4}{g} + p_4 A_4 \quad (5)$$
$$W = \rho_3 A_3 V_3 = \rho_4 A_4 V_4 \quad (6)$$
$$p_0/\rho_3 = Rt_3 \quad (7)$$
$$p_4/\rho_4 = Rt_3 \quad (8)$$

whence, by substitution and introduction of

the Mach number in the form:

$$M^2 = V^2/(g\gamma R t)$$

it can be shown that:

$$\frac{p_3}{p_4} = \frac{1+\gamma M_4^2}{1+\gamma M_3^2} \quad . \quad . \quad (9)$$

From general gas dynamic theory expressions are obtained for the velocity and total temperature:

$$V = \sqrt{\left[\frac{\gamma g R T M^2}{1+\frac{1}{2}(\gamma-1)M^2}\right]} \quad . \quad (10)$$

$$t = T/[1+\tfrac{1}{2}(\gamma-1)M^2] \quad . \quad (11)$$

whence:

$$\frac{M_3\sqrt{[1+\frac{1}{2}(\gamma-1)M_4^2]}}{1+\gamma M_4^2} = \frac{M_3\sqrt{[1+\frac{1}{2}(\gamma-1)M_3^2]}}{1+\gamma M_3^2}\sqrt{(T_4/T_3)} \quad (12)$$

$$\frac{P_3}{P_4} = \frac{1+\gamma M_4^2}{1+\gamma M_3^2}\left[\frac{1+\frac{1}{2}(\gamma-1)M_3^2}{1+\frac{1}{2}(\gamma-1)M_4^2}\right]^{-\gamma/\gamma-1} \quad (13)$$

The important expressions here are (12) and (13), and Fig. 5 summarises in convenient form the various values of the inlet Mach number and the temperature ratio.

The conclusions to be drawn from these curves are summarised as follows:

(*a*) The lower the inlet Mach number the smaller is the loss in total pressure due to combustion.

(*b*) If the inlet Mach number is in excess of 0.35, it will not be possible to exceed a temperature ratio of 2.3 before choking occurs at the end of the combustion zone. This does not mean that, assuming sufficient air is present, fuel cannot still be burnt, but it does mean that if such fuel is burnt then the pressure will rise at the outlet from the turbine accompanied by a general increase in pressures and temperatures throughout the engine.

(*c*) At the inlet Mach number of 0.5, and neglecting skin friction and "cold" losses, the maximum permissible temperature ratio would be 1.46 if an increase in engine temperatures were to be avoided. If skin friction losses were appreciable, it is probable that no heat could be added in the tail pipe without upsetting turbine outlet conditions.

Considering now the general case of the Goblin and neglecting frictional losses, for this preliminary estimate, Fig. 6 shows the thrust which would be obtained for various inlet Mach numbers. Methods for the calculation of thrust will be described later,

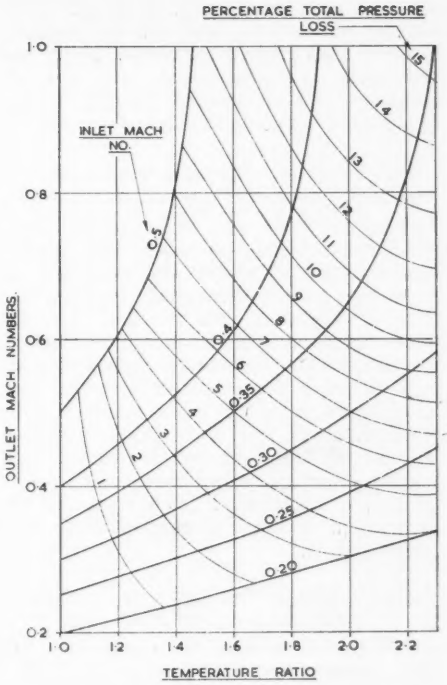

Fig. 5.

Gas flow characteristics for combustion in a parallel pipe for various inlet Mach numbers.

but for the desired increase of 30 per cent. the maximum inlet Mach number possible would be about 0.36, which would not only give choking at the end of the tail pipe, but at a temperature ratio of 2.2 would use nearly all the available oxygen. This latter aspect of combustion will be considered in more detail later. Even this condition is not wholly satisfactory, since for the particular case considered, an outlet Mach number of 1.0 combined with a loss in total pressure of 15 per cent.—obtained from Fig. 5—would give a static pressure in the nozzle of less than atmospheric—an impossible condition. The true maximum line for the Goblin is shown in Fig. 6 and indicates a maximum inlet Mach number of only 0·34.

It is clear that maximum economy would be obtained by dropping the inlet Mach number to the lowest possible value. Below 0.2, however, this involves very large tail pipe diameters, with the penalty of weight and larger aircraft cowlings. Further, the rate of proportionate increase between

$M=0.30$ and $M=0.20$ for a 30 per cent. increase in thrust, is small, and the improvement in performance does not justify the extra bulk. For this reason an inlet Mach number of 0.3 was selected for the Goblin, requiring a temperature ratio of 1.77.

2.5. Friction and diffusion losses

It is inevitable in any form of air flow ducting that losses will be incurred because of friction and eddy formation. An attempt has been made to analyse the losses through a tail pipe into its component parts of skin friction, diffusion, and so on, but the quantities involved are small and arithmetical inaccuracies assume entirely disproportionate importance. It is not felt, therefore, that such an analysis yields any great return for the work involved, but it is probably useful, before considering the overall effects of skin friction losses on performance and without going into arithmetical detail, to review the three main sources of parasitic loss in a reheat system.

2.5.1. *Skin friction losses on the walls of the tail pipe and inner cone*

The easiest method of dealing with this item appears to be the use of a general expression relating skin friction coefficient with velocity and Mach number. As quoted in Ref. (5) this is given by:

$$\frac{4fl}{D}=\frac{\gamma+1}{2}\log_e\left\{\frac{V_4}{V_3}\right\}^2+\frac{1}{\gamma}\left(\frac{1}{M_3^2}-\frac{\gamma-1}{2}\right)\times \times\left[1-\left(\frac{V_3}{V_4}\right)^2\right] \quad . \quad . \quad (14)$$

A value for the skin friction coefficient can be obtained from Generaux' equation:

$$f=0.04\ (\text{R.N.})^{-0.16} \quad . \quad (15)$$

applicable for Reynolds numbers between 4×10^3 and 20×10^6.

Values have been worked out for this expression, but the losses are exceedingly small and the work is further complicated by changes in velocity which occur when reheat is applied.

2.5.2. *Diffusion losses*

In this particular application of reheat on the Goblin the gas diffuses from the turbine outlet to the beginning of the reheat combustion system. This diffusion process takes place through a gently expanding annular passage which can be assumed to be reasonably efficient.

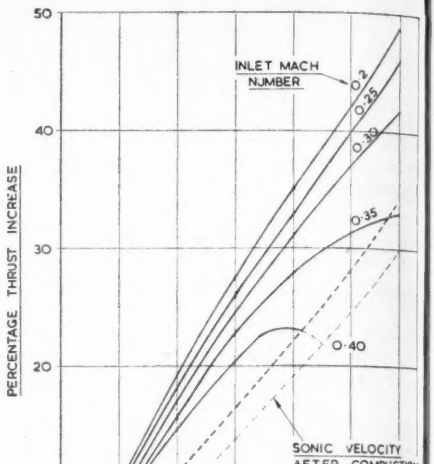

Fig. 6.

Theoretical increase in thrust of Goblin engine under sea level static conditions, neglecting cold losses.

While it is realised that test bed measurements of small quantities can easily lead to errors, experimental evidence available to the author indicates that only very small diffusion losses are actually incurred.

2.5.3. *Losses due to stabilisers*

When combustion takes place in a moving gas stream a stable flame cannot be formed unless the flame velocity is more than the gas velocity. As a general figure, flame velocities of 25-30 ft./sec. can be expected, a figure well below that of the gases in the jet tail pipe even after diffusion. This means that some form of stabiliser must be used which will allow eddies to form, thus reducing the local velocity to a figure at which continuous burning can proceed. Inevitably such a stabiliser must introduce losses.

No attempt has been made to calculate the loss in total pressure due to stabilisers, but it is felt that they form the principle source of loss.

2.5.4. *Analysis of cold losses from test results*

Since the beginning of the year (1949), when this particular investigation into reheat was started, some eight different types of stabiliser and tail pipe assembly have been tried. Original work centred on the National Gas Turbine Establishment scheme, involving the use of a number of separate conical stabilisers, each with a separate ignition unit. In later designs this was replaced by an annular stabiliser which tests showed to have good ignition and weak extinction characteristics, coupled with reasonable mechanical reliability.

In the early days tests to determine "cold" losses were made by comparing the performance of the engine at a given turbine outlet temperature, adjustments being made by altering the nozzle size. This method was clumsy, time wasting and inaccurate, and a consideration of the problem enabled a method to be devised which allowed tests to be run on a given nozzle, correction subsequently being made for changes in jet pipe temperature. In all cases losses were only considered at maximum engine speed.

The introduction of an obstruction in the tail pipe will tend to increase the pressure drop from the turbine to the nozzle for a given gas flow which, if no change is made in the propelling nozzle size, will lead to a decrease in pressure ratio across the turbine and a drop in engine speed. If the speed is to be restored, the turbine entry temperature must be raised by increasing the amount of fuel, which will not only increase the total temperature of the gases in the propelling nozzle, but may give a greater thrust.

To obtain a true conception of the cold loss, a common datum is necessary and turbine outlet conditions have again been selected. The correction to be applied to the observed thrusts to reduce them to this common level can be obtained from comparative test bed figures, with different nozzles—which is not too reliable—or it can be calculated. The latter method is probably more accurate, although it still involves the use of observed jet pipe temperatures.

For any given speed, assuming the nozzle conditions to be sub-sonic, the mass flow, static pressure in the nozzle and nozzle area are known. It is thus possible to select a range of values of T, and evaluate $(W\sqrt{T})/(A\,p)$, which leads to $V/\sqrt{T}$ and thrust $(W\,V)/g$, from which a curve of thrust against temperature can be drawn.

From test bed measurements of jet pipe temperature and thrust, with and without stabilisers and without reheat, a correction can be applied to the thrust based on the observed mean jet pipe temperature, which allows comparisons under identical temperature conditions and an evaluation of the "cold" loss.

In the calculations a conservative value of 1.5 per cent is taken, based on an analysis of test results, but it must be borne in mind that the fitting of a variable nozzle may increase this value.

2.6. Calculation of Thrust Increases

Before deciding on an inlet Mach number, or temperature ratio, for any given system, an examination of the thrust response which can be anticipated over the full operational range is essential. This entails the calculation of the total pressure at the propelling nozzle for various values of the above quantities, due allowance being made for variation in cold and momentum pressure losses.

To establish a datum for this work involves the calculation of the total pressure in the tail pipe for normal conditions without reheat. For this, a knowledge of the engine air and fuel mass flows, thrust, and turbine outlet total temperature at each condition is required, when it becomes a fairly simple matter to obtain the datum total pressure (equations (16) and 18)). Application of the fixed cold loss correction and the variable momentum loss correction, obtained from Fig. 5, will then give the required total pressure with reheat in operation.

Both sub-sonic and choking conditions must be considered, and expressions have been derived in the next section which give the proportionate thrust increase, assuming the nozzle to be sub-sonic before and after the application of reheat, or choking before and after reheat.

It is possible to obtain conditions where the nozzle is choking before the application of reheat but sub-sonic afterwards; no general formula is given for this case, although it can be worked out from first principles. It only covers a narrow range of conditions, however, and a detailed analysis is hardly justified.

2.6.1. *Sea level static conditions (sub-sonic)*

In this case, the static pressure in the nozzle is atmospheric and the value for

$$(W_a+W_f)\sqrt{T_5}/(A\ p_5)$$

is immediately available for normal conditions without reheat. The application of the appropriate curves, or directly by the application of the equation (16), gives the value for the pressure ratio and hence the total pressure upstream of the nozzle.

$$\frac{(W_a+W_f)\sqrt{T_5}}{A_5\ p_5}=1.645\frac{[1-(p_5/P_5)^{(\gamma-1)/\gamma}]}{(p_5/P_5)^{(\gamma-1)/\gamma}} \quad . \quad . \quad . \quad . \quad (16)$$

where $\gamma=1.33$

As already explained, the application of a cold loss and momentum loss correction will now give the total pressure at the nozzle with varying degrees of reheat.

It can be shown that the proportionate thrust increase is now given by the expression:

$$\frac{F'}{F}=\frac{(W_a+W_f')}{(W_a+W_f)}\frac{V_5'/\sqrt{T_5'}}{V_5/\sqrt{T_5}}\sqrt{(T_5'/T_5)} \quad (17)$$

where $V_5/\sqrt{T_5}$ = thrust factor, and is obtained from the pressure ratio by the equation:

$$V_5/\sqrt{T_5}=\sqrt{[2\,g\,R\,\gamma/(\gamma-1)]}\times\times[1-(p_5/P_5)^{(\gamma-1)/\gamma}]$$

To simplify calculations, the thrust factor has been shown in graphical form against the pressure ratio for various values of γ in Fig. 8.

A comprehensive series of thrust curves has been worked out for the Goblin III and is presented in Fig. 7. These are similar to those in Fig. 6, but the proportionate thrust increase is slightly lower for a given temperature ratio. For the purpose of this curve, a cold loss of 1.5 per cent. has been assumed.

2.6.2. *Flight conditions (choking)*

In this case, the thrust is made up of a velocity component and a pressure component. Under these circumstances the total pressure will have to be calculated over the flight range and equation (18) enables this to be done from a knowledge of the engine

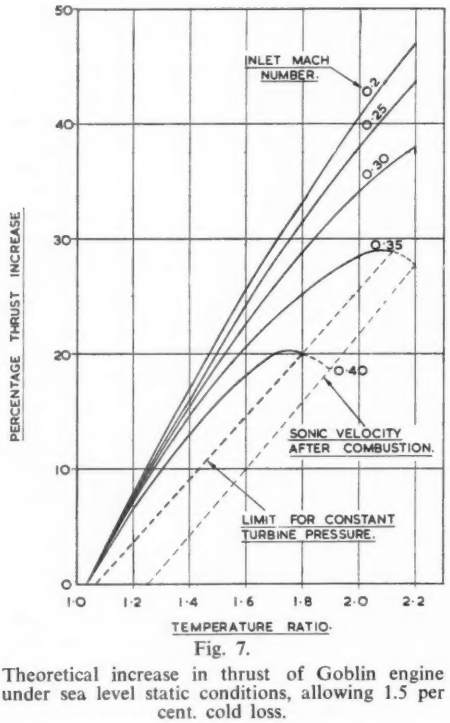

Fig. 7.

Theoretical increase in thrust of Goblin engine under sea level static conditions, allowing 1.5 per cent. cold loss.

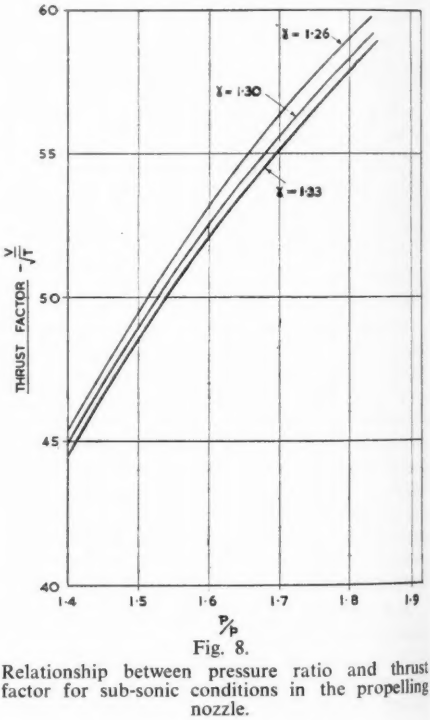

Fig. 8.

Relationship between pressure ratio and thrust factor for sub-sonic conditions in the propelling nozzle.

thrust, mass flow, forward speed and nozzle size, for normal conditions without reheat. For $\gamma = 1.33$,

$$P_5/p_a = 0.798\ \{1 - F_{gross}/(A_5\, p_a)\} \quad (18)$$

As before, application of cold loss and momentum loss corrections will give the total pressure at the nozzle and the proportionate thrust can then be calculated from the following equation:

$$\frac{F'}{F} = \frac{(W_a' + W_f')}{(W_a + W_f)} \times \frac{\left[\frac{1}{\sqrt{[\gamma'(\gamma'+1)]}}\left\{(\gamma'+1) - \left(\frac{p_a}{P_5'}\right)\left(\frac{\gamma'+1}{2}\right)^{\frac{\gamma'}{\gamma'-1}}\right\}\right]}{\left[\frac{1}{\sqrt{[\gamma(\gamma+1)]}}\left\{(\gamma+1) - \left(\frac{p_a}{P_5}\right)\left(\frac{\gamma+1}{2}\right)^{\frac{\gamma}{\gamma-1}}\right\}\right]} \times \sqrt{\frac{T_5'}{T_5}} \quad . \quad . \quad (19)$$

where Thrust Factor $=$

$$\frac{1}{\sqrt{\gamma(\gamma+1)}}\left\{(\gamma+1) - \left(\frac{p_5}{P_5}\right)\left(\frac{\gamma+1}{2}\right)^{\frac{\gamma}{\gamma-1}}\right\}$$

It has been found that the effect of changes in γ between 1.33 and 1.26 on the proportionate thrust calculated by equation (19) is negligible and the thrust factor for choking conditions has therefore been shown in graphical form against pressure ratio for $\gamma = 1{\cdot}33$ in Fig. 9.

Under most flight conditions the nozzle is choking, and it was felt that the effect of reheat at varying forward speeds could best be illustrated from the application already quoted, namely 30 per cent. reheat on the Goblin. Changes in altitude affect the proportionate increase slightly, but in general this is less than 2 per cent., and its effects, therefore, have been averaged out. Fig. 10 thus shows the variation in the proportionate thrust increase for a given temperature ratio with forward speed, at all altitudes up to 50,000 ft.

2.7. Final Nozzle Diameter

The dimensions of the tail pipe have now been determined, and a forecast has been made of the thrust increases which will be obtained, provided that the gas temperatures are increased in a certain ratio. The only important dimension remaining to be fixed is that of the nozzle size.

It is not correct to use curves such as in Ref. 1 for this work since the temperatures of the gases passing through the final nozzle are so high that both the gas constant and the ratio of specific heats may have altered

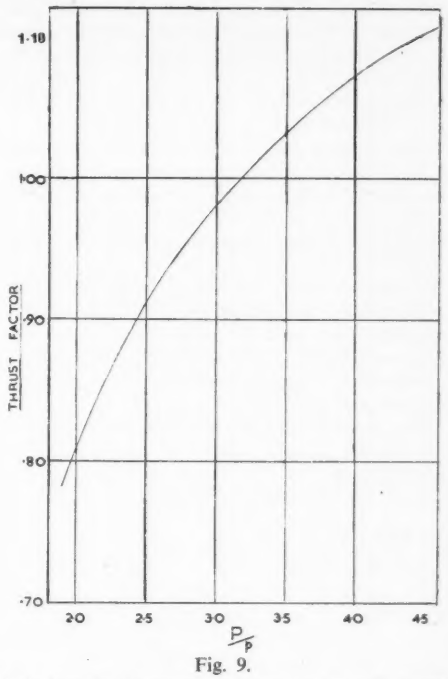

Fig. 9.

Relationship between pressure ratio and thrust factor for choking conditions in the propelling nozzle.

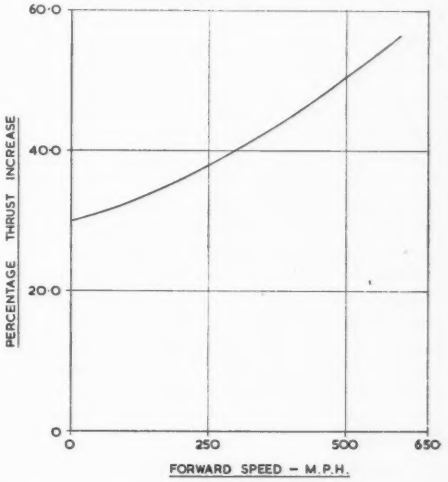

Fig. 10.

Effect of forward speed on proportionate thrust increase of Goblin at all altitudes from sea-level to 50,000 ft.

materially. Figs. 11 and 12 show the variations in the gas constant and the ratio of specific heats against overall fuel/air ratio, together with a graph showing the maximum temperatures which can be anticipated for a given overall fuel air ratio.

The overall fuel/air ratio, rather than the reheat fuel/air ratio, has been used since the heating process is actually a continuous one in which the gases are heated from atmospheric conditions to turbine outlet conditions in the engine, and subsequently reheated to the final nozzle conditions. No work has been taken from the gases, if radiation losses are neglected, and the whole heat input has been converted to thermal and kinetic energy irrespective of any internal exchanges between turbine and compressor.

No attempt has been made to generalise as regards nozzle size, but equations (20) and (21) enable the nozzle diameter to be calculated for any specific case.

(*a*) *Sub-sonic*

$$\frac{(W_a' + W_f')\sqrt{T_5'}}{A_5' p_a} = \sqrt{\left[\frac{\gamma'}{\gamma'-1}\,\frac{2g}{R}\right]} \frac{\left\{1 - \left[\frac{1}{\left(\frac{P_5'}{p_a}\right)}\right]^{\frac{\gamma'-1}{\gamma'}}\right\}^{\frac{1}{2}}}{\left[\frac{1}{\left(\frac{P_5'}{p_a}\right)}\right]^{\frac{\gamma'-1}{\gamma'}}} \quad (20)$$

(*b*) *Sonic*

$$\frac{(W_a + W_f')\sqrt{T_5'}}{A_5' P_5'} = \sqrt{\left(\frac{\gamma' g}{R'}\right)}\left[\frac{2}{\gamma'+1}\right]^{\frac{\gamma'+1}{2(\gamma'-1)}} \quad (21)$$

A point worth noting is that since the turbine outlet temperature remains substantially constant on a normal engine for any given diameter of nozzle, and a constant temperature ratio and a constant proportionate cold loss have been assumed, then for choking conditions the value W/P_5' will remain constant, and there will be no need to change the nozzle diameter for differing conditions of flight. This should not be confused with the other aspect of reheat, where varying degrees of thrust increase are required, in which case variations in nozzle size over the complete range will be necessary.

3. FUEL CONSUMPTIONS

Fuel consumptions can be obtained by application of Fig. 12 which shows temperature against overall fuel/air ratio. The

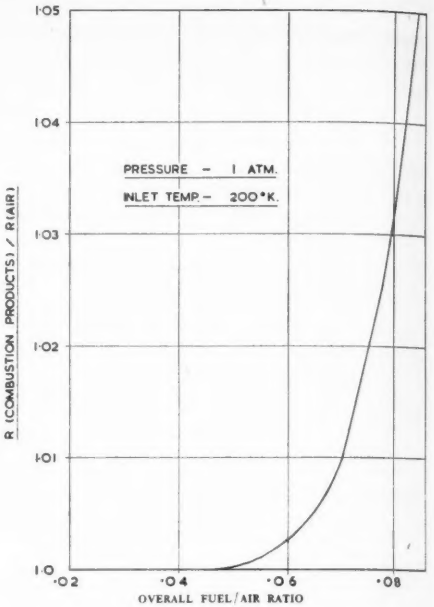

Fig. 11.
Variation in R for various overall fuel/air ratios.

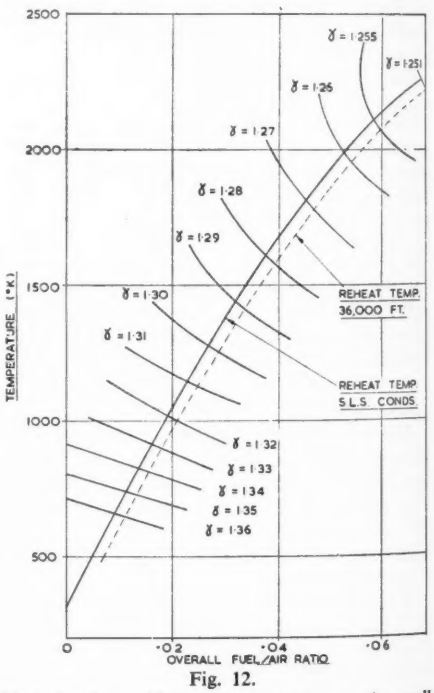

Fig. 12.
Variation in γ with reheat temperature and overall fuel/air ratios.

general tendencies follow directly from the preceding calculations, whence it will be seen that maximum economy can only be obtained if losses are kept to a minimum, and combustion efficiencies are kept high. High losses will mean excessive temperature ratios for a given thrust, while the fuel losses due to poor efficiency need no explaining.

It has not been thought worth while to try and evolve a general expression for fuel consumption, since figures can be obtained so easily from the curve. At the same time it is interesting to obtain some idea of the general trend, and the following curves applicable to the Goblin are provided:

Figure 13. Reheat fuel flow per pound of air for various temperature ratios and combustion efficiencies, assuming a turbine outlet temperature of 1,000° K.

Figure 14. Approximate curves showing variations in specific consumption at various speeds and altitudes with changes in the amount of reheat.

Figure 13 will apply very closely to most present-day applications, and provides a convenient way of measuring combustion efficiency, from a knowledge of thrust and fuel consumption.

4. COMBUSTION

4.1. General

The problem of reheat combustion involving the burning of large quantities of fuel in a fast-moving gas stream is far from easy. If the system is to function smoothly, a stable flame must be formed which will burn steadily over a wide range of mixture strengths and mass flows; it must be easy to ignite and combustion must be efficient. Further, the method of flame stabilisation should avoid high losses when the reheat system is not operating, since these losses involve a decrease in normal operational thrust and an increase in fuel consumption.

It is difficult to be specific about the actual combustion process. Many factors are involved: the degree of preheating of the fuel, the amount of vaporisation prior to ignition, the composition and homogenity of the combustible mixture, the amount of inter-diffusion of the fuel and air, the stability of the flame, and the effect of stabiliser design

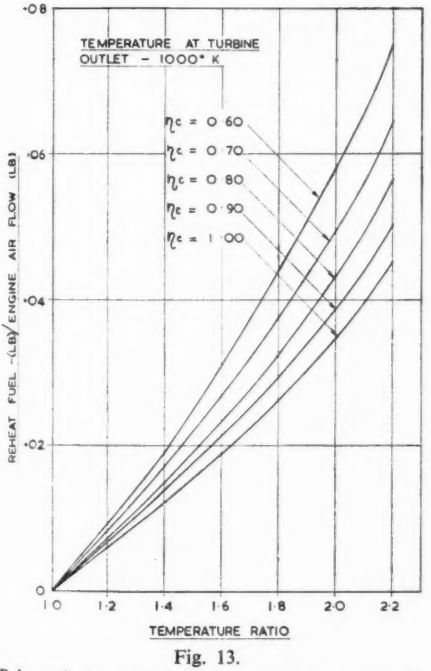

Fig. 13.
Reheat fuel consumption per lb. of air through engine.

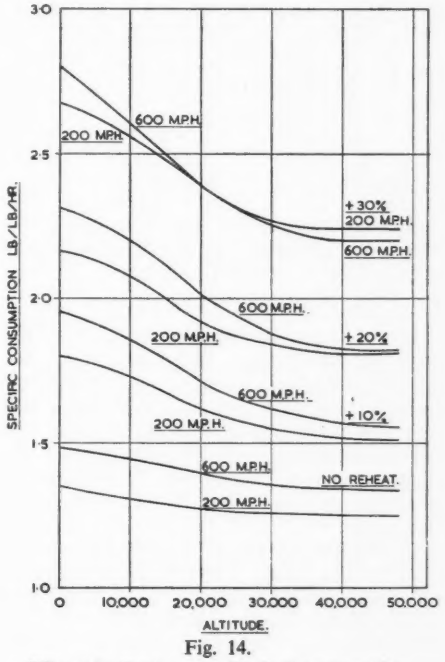

Fig. 14.
Effect of reheat on specific fuel consumption.

on rich and weak limits, the amount of control required over the reheat combustion, and the conditions under which it is required to ignite and operate.

Information gleaned from many reports indicates a conflicting state of affairs. Briefly:

(*i*) The high temperatures at which combustion is initiated will aid vaporisation, widen the stability limits and tend to produce a short flame. This is a direct consequence of a reduction in the delay, or pre-combustion reactions. Theoretical considerations of the time lag do not appear to correlate very well with practice, the actual time being much less than the theoretical.

(*ii*) The high proportion of nitrogen in the gases, together with the presence of CO_2 and H_2O tends to slow the reactions, increase the flame length and narrow the stability band. These effects could be appreciable for the mixture under consideration, but the effects of high temperatures tend to act in the opposite direction and the indications are that the final combined effect will not be serious.

(*iii*) The effect of velocity will be to concentrate the fuel around the stabiliser. With increasing amounts of reheat fuel, it is to be anticipated that the rich extinction point will occur earlier at high velocities than at low. This is borne out in practice.

Summarising, it is impossible to lay down any hard and fast rules about stabiliser design or methods of fuel injection. The only practical approach to this part of the problem is that of trial and error.

4.2. Ignition

In view of the high temperatures of the gases in the tail pipe, it is tempting to try and ignite the mixture spontaneously. Everything is in its favour, in that the air and fuel are well mixed, the fuel should be vaporised by the time it reaches the stabiliser and, under sea level conditions, a pressure of 26-30 lb./sq. in. will exist in the tail pipe.

Considerations of this problem, however, indicate that under atmospheric conditions, although cool flames form at temperatures up to 700°C., no true combustion will take place until some 800°C. is reached. Given a sufficiently low velocity in the tail pipe and an ideal air/fuel ratio, it is possible that the heating effects of the early stages of combustion may increase mixture temperatures locally to ignition point, when inflammation of the whole mixture will occur.

If the possibility of spontaneous ignition at take-off conditions is granted, it is unlikely that it would take place at altitude where the maximum absolute tail pipe pressure would not exceed 6 lb./sq. in. Further, the range over which the mixture would ignite would be so small as to preclude its operational use—it would be too easy to miss the ignition point.

Some form of spark ignition is essential, therefore. Information on the ignitability of fuel/air mixtures at high temperatures and low pressures is practically non-existent, but the tendencies will obviously be the same as for ignition in ordinary combustion chambers. On the average it appears that a high spark energy is necessary if a combustion chamber is to light at extreme altitudes. With the higher pressures and much higher temperatures pertaining in the reheat system at the instant of ignition, this energy requirement will be less, but pending accurate data on the subject it appears unwise to assume that the normal booster coil will be sufficient.

4.3. Fuel Injection

It appears to be fairly well established that fuel injection for reheat can be carried out quite satisfactorily, using plain jets. Atomisation in this case is a function of the gas velocity, the jet size and the fuel pressure, and tests have shown that quite fine sprays are possible. The major disadvantage appears to be localisation of the fuel in the body of the air, giving rise to uneven fuel distribution in the jet pipe. This can be overcome, to some extent, by positioning the jets well upstream of the stabiliser, but under these circumstances radial localisation of the inflammable mixture is difficult and ignition will occur only when quite a large proportion of the gas in the jet pipe is above the weak limit, implying a sudden rise in thrust, with no control, at low thrust boosts. This may be acceptable in certain cases, but shock loads of any kind are, in general, better avoided.

4.4. Combustion Zone Volume

The easiest way of determining this quantity is from a consideration of combustion intensity. The usual criterion for

this is the heat input per atmosphere per hour per cubic foot, as given by the expression:

$$I = \frac{W_f \text{ (lb./hr.)} \times \text{C.V. (C.H.U./lb.)}}{P \text{ (atmos.)} \times \text{Volume (ft.}^3\text{)}} \quad (22)$$

No correction will be necessary for altitude and forward speed, other than that which may be caused by changes in combustion characteristics due to low pressures, since, as has already been shown (Section 2.7), the ratio W/P is constant for all conditions of flight when the nozzle is choking. A consideration of the combustion intensity shows that it can be written in the form:

$$I = (\text{Reheat Fuel/Gas Ratio}) \times \times \frac{W_a}{P_5'} \times \frac{\text{Calorific Value}}{\text{Volume}} \quad . \quad (23)$$

which, if the weight of the reheat fuel is neglected, is constant for any given system.

A very approximate method of determining the minimum combustion zone diameter involves an assumption of effective flame velocity.

Imagine that the flame is stabilised at a number of points, situated on the mean pitch circle of the tail pipe, that a homogeneous combustible mixture is evenly distributed around the stabiliser system, and that the flame is stable and self-supporting, and radiating into the main stream from the pilot zone. An accurate flame velocity is difficult to assess, bearing in mind the general turbulence of the gases, the high temperature and the presence of carbon dioxide, water vapour and an excess of nitrogen, but a figure of 30 ft./sec. is probably not unreasonable.

In still air, an explosion flame front would move spherically away from the source; in a high speed gas stream it would move radially at a fairly slow speed—the flame speed—and downstream at the gas speed plus the flame speed.

The actual process in the reheat zone will be a little different. Some measure of recirculation is essential if the flame is to be self-supporting, and the shape of the flame is envisaged as a hot central core—the piloting zone—surrounded by a conical envelope of flame. The solid angle of this envelope will be determined by the resultant of the gas velocity and the flame velocity.

A number of trial calculations indicated that the volume obtained on this basis would give a longer tail pipe than is forecast by considerations of combustion intensity. The implications of this are that if the tail pipe is made shorter than is called for on a basis of flame speed, the flame will not reach the sides, and all the fuel must be burnt in an inflammable core. The difficulties of controlling the injection of fuel to give a clearly defined boundary of this nature are obvious, since the outer layers must be below the inflammable limit and the fuel therein must be wasted. It appears inevitable, therefore, that low thrust boosts, to some extent, must be inefficient, a conclusion which is borne out by test results.

5. GENERAL DESIGN REQUIREMENTS

The reheat system on the Goblin has been designed with the following specific objects in view, although not all have yet been achieved:

1. Good ignitability under all conditions of flight, the reheat lighting up gently and without shock, subsequently being opened up smoothly to full thrust.
2. Wide stability limits, with special emphasis on the weak end, to ensure smoothness and pilot's control up to maximum thrust.
3. Low cold pressure loss due to stabilisers, to give approximately normal operation when reheat is not in operation.
4. Reasonable combustion pressure loss and efficiency, consistent with a realistic tail pipe which can be installed in existing types of single-engined fighter.
5. Good mechanical reliability and cool operation of the tail pipe and propelling nozzle.
6. Variable degree of reheat, under the control of the pilot, this to be interconnected with a variable nozzle to ensure optimum conditions at all times.

The method adopted for flame stabilisation and fuel supply is quite orthodox. A V-shaped ring is installed in the tail pipe to act as a stabiliser, fuel then being sprayed into the hot gases upstream of this ring, the combustible mixture so formed eddying around the stabiliser and giving a self-piloting flame, with good stability characteristics. A ring stabiliser was selected as offering the best conditions for light-up and, in fact, it has been found that although two plugs were fitted, one was sufficient to give smooth ignition, with a thrust increase of only some 100 to 150 lb.

No variable nozzle was fitted, since the

design of this component presented many difficulties which have not yet been overcome. The amount of thrust increase, therefore, was dictated by the thrust available without reheat which had to be sufficient for take-off. From a combustion point of view, this was not a serious drawback, since the performance could be examined just as easily without a variable nozzle; from a light-up point of view, it was a handicap, since it reduced both the temperature and the pressure in the tail pipe.

6. TEST TECHNIQUE AND ANALYSIS OF RESULTS

In the evaluation of a reheat system, the usual two stages of any development must be covered, namely, preliminary tests on the ground, augmented if need be, by rig tests, and comprehensive flight trials.

In the preceding sections a detailed theory has been put forward which has attempted to cover all the important variables and to forecast, from the engine characteristics and the dimensions of the reheat system, the performance which will be obtained. In all these calculations one vital unknown quantity can only be found by experiment, namely, the combustion efficiency, a factor which has a direct and major influence on the fuel consumption, and an indirect, but nevertheless vitally important influence on the extinction limits.

There is a certain amount of controversy regarding the exact definition of combustion efficiency in this application, but the author feels that little ambiguity is really possible. In all the work which has been done so far the reheat combustion efficiency has been taken as the ratio of the fuel ideally required to heat the gases to the nozzle total temperature, to the fuel actually used, due allowance being made, if necessary, for dissociation.

The following paragraphs, therefore, are devoted to a detailed study of the methods of analysis used and the degree of correlation which has been reached between theoretical and actual results.

6.1. GROUND TESTS

It is assumed, as has already been stated, that the air mass flow through the engine is known, within close limits. The normal engine data will have to be recorded during tests, namely, engine speed, thrust, jet pipe temperature, engine fuel consumption, and so on, and in addition, the turbine outlet temperature and pressure.

As was stipulated earlier, the datum conditions for any comparative measurements are the engine speed—*i.e.* the air mass flow—and the turbine outlet temperature. While accurate setting and measurement of engine speed is a comparatively simple operation, the setting and measurement of the turbine temperature is most uncertain, involving the determination within very close limits of the mean temperature of a gas stream some two sq. ft. in area, travelling at over 1,000 ft./sec. As a matter of fact, this proved almost impossible; variations of over 100°C. between thermo-couples were experienced and were further complicated by changes in the reading of individual couples with different installations.

In the early days of this investigation, this handicap was accepted, it being hoped (although the hope was subsequently found to be misplaced) that the mean value of the couples would be reasonably near the true value, and that the patternation across the jet pipe would remain sensibly constant. A modification of accepted technique was the use of the turbine outlet temperature as the datum, as against the usual nozzle temperature, and tests were run on the standard engine to determine the relationship between these two measuring points. Theoretically, they should be the same, but in actual practice the thermocouples indicated that the turbine outlet was apparently hotter than the normal jet pipe temperature. Allowance was made for this in subsequent tests.

In the early work, efforts were concentrated on shortening the flame and obtaining good stabilisation and light-up characteristics. Fig. 15 shows the flame lengths obtained at first, although even here it will be seen that the tail pipe is cool. The thrust increase was not great.

The original system involving individual stabilisers was soon dropped and a ring stabiliser adopted for reasons of stability and simplicity, especially as regards the light-up. Many methods of fuel injection were tried, but ultimately a system was evolved which was considered functionally reasonable, and designs were proposed to enable a Vampire to be used as a flying test bed. In the meantime a series of bench tests were put in hand using three different nozzle sizes to obtain a range of operation up to 20 per cent. thrust increase.

Accurate calibration of the standard engine was first undertaken to determine a datum

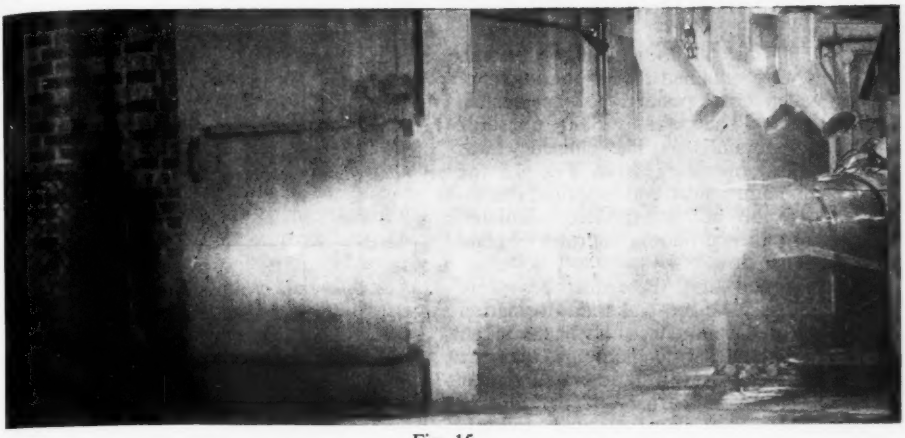

Fig. 15.
Early design of reheat system showing excessive flame length.

thrust, fuel consumption and turbine outlet temperature. This test was then repeated with a prototype reheat tail pipe similar to that which would be used in flight, and the cold losses evaluated by the method described in Section 2.5.4, which unfortunately relies on consistency of temperature recording. In the case under consideration cold losses were found to be less than one per cent., but on all the performance estimates which have so far been made, a loss of 1.5 per cent. has been assumed.

Performance checks were now taken with reheat in operation. Instrumentation comprised four equally spaced thermocouples at the turbine outlet position and means for measuring the total fuel flow, the reheat and engine flows being separated afterwards from a knowledge of the burner pressures and flow numbers. This method was found to be quite accurate, provided that the pressure gauges were carefully checked and read. Figures were then taken at various reheat fuel flows up to the maximum permissible turbine temperature and curves drawn of thrust, engine fuel flow and jet pipe temperature against reheat fuel flow.

A number of such tests were made with three nozzles, differing in diameter by $\frac{1}{4}$ in., which gave three points on the general temperature thrust curve.

Preliminary analysis, based on thermocouple readings gave very erratic results, and it soon became obvious that some other datum would have to be chosen. The obvious one was turbine outlet pressure, but unfortunately this had not been taken in every test, and many retests would have been necessary. On the other hand, accurate figures of engine fuel consumption were available and since it was a reasonable assumption to say that at a given engine speed and fuel flow temperatures would be identical, engine fuel flow was selected as a datum and thrusts were compared at equal engine consumptions. All thrusts, consumptions and temperatures were corrected for atmospheric conditions in the usual way, for all conditions both with and without reheat.

While this basis of comparison was accurate for thrusts, it did not give the necessary data to calculate combustion efficiency. From a knowledge of the thrust, mass flows—air and fuel—and nozzle size, however, the necessary temperatures could be calculated. The method was to assume a value of T_5', and evaluate the quantity $(W_5' \sqrt{T_5'})/(A_5' p_a)$. The jet velocity could then be calculated from the thrust $F = W_5' V_5'/g$, whence, from the assumed value of T_5 a value of $V_5'/\sqrt{T_5'}$ was obtained, which in turn gave another value of $(W_5' \sqrt{T_5'})/(A_5' p_a)$. By successive approximations, it was then possible to arrive at a value for the gas temperature which gave the same value of

$$(W_5' \sqrt{T_5'})/(A_5' p_a)$$

irrespective of the method of approach. In the range of temperatures covered by the tests no allowance was made

for changes in γ, as this made a negligible difference.

Applying this method throughout, temperatures were obtained for all conditions of reheat which corresponded to the normal engine speed. The points on Fig. 16 show the results obtained. The efficiency was calculated by obtaining from Fig. 12 the ideal engine and reheat consumption, corresponding to the aforementioned calculated temperature, thence finding the ratio of ideal to actual fuel flow.

It was found that these methods were consistent and gave most of the information required from ground tests. There is an alternative method of obtaining efficiency from a knowledge of the static pressures, but this has not yet been tried and preliminary calculations indicate that for small amounts of reheat the static pressure ratio would be low, calling for very accurate measurement. It did not appear necessary on the ground where thrusts were available; in flight, however, it provides an alternative method of obtaining the information, and is described in more detail in Section 6.2.

6.2. Flight tests

Detailed investigation into the general behaviour of reheat at altitude and varying speeds can only be made in Great Britain by installing it in an aeroplane. In such a flying

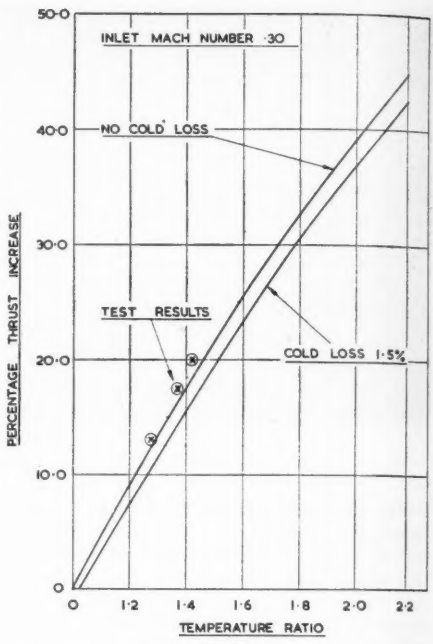

Fig. 16.
Correlation of test results and theoretical thrust increase.

Fig. 17.
Vampire VV 454 modified for reheat.

test bed, details of aircraft handling, engine speed and temperature control, and so on, can, to a certain extent be left to the pilot, thus eliminating the complications of extra automatic devices, and any deviations from "standard" conditions can be subsequently ironed out in the analysis. The main limitation is the provision of adequate thrust for take-off without reheat, since the variable nozzle, which would allow normal engine thrust to be obtained, has not as yet been well enough developed to fly.

With these points in mind, and also considering the availability of aircraft and parts, a Goblin-engined Vampire was modified to incorporate a reheat system of the type described in the preceding sections. Photographs of the aeroplane and reheat system are shown in Figs. 17 and 18. The fuel system was of the simplest, comprising two Lucas pumps supplying fuel through two branches, one going to the normal engine control box, and the other to a specially designed reheat shut-off cock and control box, operated by a single lever in the pilot's cockpit.

Flight clearance was obtained on completion of a 25-hour special category test, and installation and preliminary clearance flights were completed without incident. No special instrumentation was incorporated for this early work other than a reheat burner ring pressure gauge, since it was felt that some idea of the potentialities and limitations of the scheme should be obtained first. At this stage nothing was known as to ignition properties in flight, aircraft performance with reheat, whether satisfactory take-offs were possible, reheat rich and weak stability at altitude, and so on, and until the operational limits were determined, a general survey seemed best. The results quoted in the ensuing paragraphs, therefore, apply only to this preliminary work.

General performance proved to be good. As indicated by the lowest test point in Fig. 16, a thrust increase of 13 per cent. at sea level was obtained, which would increase to nearly 20 per cent. at 400 m.p.h. and 25 per cent. at 600 m.p.h. In practice these thrust gains resulted in an increase of over 40 m.p.h. in maximum forward speed and a 30 per cent. reduction in the time to climb to operational height. Stability was excellent, there being no sign of rich extinction when the air flow was cut down by reducing the engine speed, and weak extinction only occurred when the fuel was shut off altogether. Aircraft handleability was affected to a certain extent, since, in order to get the extended tail pipe into the Vampire, it had been necessary to tip it upwards by 6° from the longitudinal axis of the engine, and this gave a nose-up effect which altered the trim. As far as reheat was concerned, there was a slight vibration, but

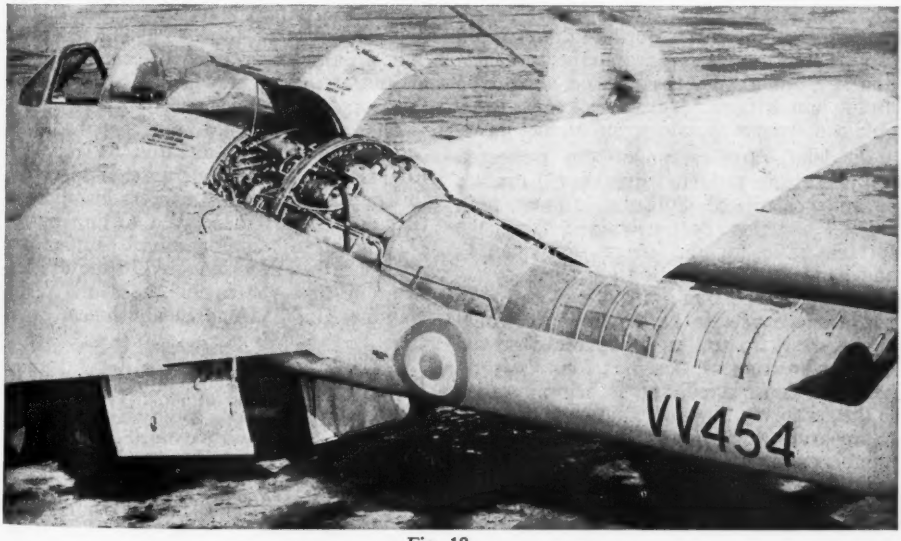

Fig. 18.
Vampire VV 454 with engine cowlings removed.

otherwise the aircraft was normal; examination after some hours' operation with reheat in use did not indicate that the tail pipe had over-heated in any way.

The characteristic in which most interest was centred was the combustion efficiency. This could be calculated from the engine and reheat burner ring pressures and a knowledge of the air flows at the various flight conditions. From this information and the burner flow numbers, the fuel/air ratio in the engine could be obtained, and hence, from Fig. 15, the turbine outlet temperatures, either assuming some value for the engine combustion efficiency or, since it will certainly be high, assuming it to be 100 per cent. If the engine speed does not change, then it can be assumed, as a first approximation, that the nozzle temperature is constant, irrespective of forward speed and altitude, enabling an estimate of the final temperature to be made from a knowledge of ground level conditions. Fig. 12 will then give an overall fuel/air ratio, from which the ideal reheat fuel flow is obtained. The actual fuel flow is known from the reheat burner ring pressure, and hence the combustion efficiency can be determined.

As already mentioned there is another way of obtaining combustion efficiency from a measurement of the static pressure drop in the parallel section of the reheat combustion chamber. It suffers from a number of fundamental snags, in that it assumes no frictional losses and is only applicable in a duct of constant area. Further, the pressure drop is small and very accurate instrumentation is essential, but the method should be useful and will provide a check on results obtained from a measurement of fuel flows.

In detail, application of the measured static pressure ratio to equation (9) enables M_4 to be calculated. Substitution in equation (12)—T_3 having been obtained from the fuel/air ratio for the engine—provides T_4 and hence, the combustion efficiency.

It was realised that if a full exploration of performance was to be obtained, it would be necessary to use an automatic observer. Such an instrument, designed for this work, has been constructed, and is being installed in the aeroplane. Instrumentation is designed to measure fuel flows from burner flow numbers and pressures, combustion efficiency from static pressures and fuel flows, and the complete aircraft performance. A steady increase in the nozzle size—possible now that take-offs with reheat have become normal—is envisaged, giving comparative performances at various degrees of reheat and allowing a complete picture to be built up of the practical possibilities of the system.

7. CONCLUSIONS

From the preceding work the inherent possibilities of the reheat system are clear. It can provide proportionate increases of thrust of up to 45 per cent. for take-off, and up to 85 per cent. for combat at 600 m.p.h. The specific fuel consumptions will rise to a value of some 3.0 lb./hr./lb. for the above conditions, but when this is combined with the improved rate of climb, assuming good combustion efficiencies, it is found that the amount of fuel required to reduce the time taken to reach operational altitudes is not excessive.

The chief disadvantage seems to be that the installation of stabilisers and a variable nozzle will decrease the tail pipe efficiency under normal conditions and the use of very high temperatures—some 2200°K.—will make the mechanical construction of a variable nozzle a difficult undertaking. Weight increases will also have some effect, although these will not be great. There seems to be no reason why any reheat system should increase the installation weight—which is taken to include extra aircraft fairings—by much more than 20 per cent. of the engine weight. Since this gives a small engine with a maximum power which could only be equalled by a much larger power unit, without the necessity of carrying around all the extra weight when it is not wanted, this does not seem a high price to pay.

The author would like to thank the Ministry of Supply and the de Havilland Engine Co. Ltd., for permission to publish this paper, and also to express his appreciation of the assistance given to him by Mr. D. Murkett in its preparation. Further great assistance has been rendered in the development of this system by the work of the National Gas Turbine Establishment.

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DESTRUCTIVE ENERGY IN AIRCRAFT PRESSURE CABINS

by

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1. INTRODUCTION AND SUMMARY

1.1. The energy that is suddenly released when a structure fails under load usually presents one of the minor problems in strength testing technique. Care has to be taken to ensure that the strain energy in the structure does not do further damage after primary failure and so obscure the location of the original weakness; and the possibility of risks to personnel must not be overlooked. As a general rule, however, aircraft structures fail in strength tests without untoward effects. Most of the strain energy within the structure is absorbed internally or dissipated in loading gear and anchorages, and the most noticeable effect is the noise and vibration which is produced.

The strength testing of aircraft pressure cabins presents a much more serious problem if air is used. The energy of the compressed air is many times greater than the strain energy in the cabin structure and, furthermore, it is released in such a way that each unit of energy is likely to do more damage.

Failure of a pressure cabin in an air test, in fact, frequently amounts to an explosion. The risk of damage to buildings and testing equipment is serious. Still more serious is the risk of injuring personnel. Both risks may be greatly reduced by protective arrangements, but there still remains the damage done to cabin structure. Frequently this disintegrates completely and leaves no evidence of the original weakness.

An alternative is to use water for reproducing air pressure. Recent work has shown conclusively that this method is satisfactory for most purposes, so far as structural strength is concerned. At the working pressures used water is virtually incompressible and the energy released is minute, compared with that for compressed air. It is necessary only to compensate for the dead weight of the water, which is done automatically by submerging the entire structure in water.

1.2. It is generally realised that the bursting of an air-filled pressure cabin under test is dangerous and destructive. It is also quite well known that water testing is the practical way of overcoming most of the difficulties. Information concerning the comparative destructive effects of air, water and the structure, however, is not readily available.

The primary purpose of the work described in this paper is to obtain numerical values for the mechanical energy released in the three forms mentioned, for the full range of conditions likely to be encountered. These figures, when taken alone, are a rough but useful guide to the comparative destructive effects to be expected.

It is necessary, however, to pay at least some general regard to the extent to which any given amount of released energy is destructive. The mechanism of destruction is an involved subject for which precise discussion is beyond the scope of this paper, but it is possible to consider certain points broadly and qualitatively; and the following are selected as being the most significant.

With structural strain energy, much of the energy released on failure does little, if any, harm. A large proportion of the energy is absorbed internally as vibration which soon dies away. Energy is also transmitted to the supports, to be likewise dissipated.

The energy stored in compressed air, on the other hand, is released outward, and pressure forces tend to continue after the structure has ceased to resist them. The destructive effect, moreover, may be increased by the formation of pressure waves, so that while the average pressure

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falls on expansion, local pressures at the wave crests remain considerably higher than the average.

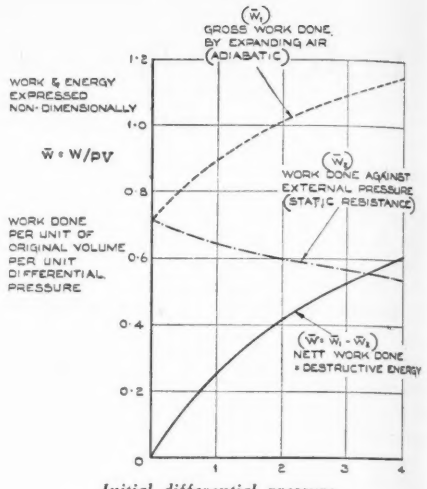
The behaviour of water is complicated by the inertia, not only of the water which is compressed inside the cabin, but also of the water in which the pressure cabin is submerged. The effect is to localise the disturbance and leads to appreciable energy absorption without much damage being done.

1.3. As to the calculations of available energy here undertaken, those for water are straightforward. Values obtained for the destructive energy of compressed air, however, depend upon basic assumptions concerning what energy is available for destructive purposes. This is probably the reason why different workers in the past have sometimes obtained different results. In this paper the aim is to choose what appear to be the best assumptions and to state them clearly so that they may be queried.

The main difficulty in determining the strain energy of the structure is that of obtaining results that are broadly applicable to all practical designs of pressure cabins. Idealised forms of pressure cabins based on spheres and cylinders are first investigated. From consideration of the results obtained it is then decided that these have a wider application than might have been expected. It is considered necessary, however, to bear in mind the unavoidable departure from the ideal condition assumed for purposes of calculation. It is concluded that the ideal figures give an over-estimate of the strain energy in a real structure, since lower structural efficiency leads to lower average stresses and hence, to lower strain energy.

1.4. The more important results of this investigation are set out in diagrams (Figs. 2 and 3) where the various forms of released energy are compared. Reference to these curves may be necessary in particular cases since results depend upon working pressure. It is possible, however, to obtain rough general conclusions as to the comparative amounts of destructive energy on the assumption that the working pressures lie between one and two atmospheres.

On this basis the first conclusion reached (*see* Fig. 2) is that the available energy of air is about *ten thousand* times greater than that of water in a corresponding test. This figure alone establishes the superiority of water testing over air testing, from the point of



$$Q = \frac{\textit{Initial differential pressure}}{\textit{External air pressure}}$$

Fig. 1.
Curves for work done by expanding air.

view of damage, without even considering the further advantages of water arising from its inertia and capacity for energy absorption.

On the same basis the available energy of air (Fig. 3) is seen to be about *fifty* times greater than the structural strain energy as calculated for the idealised structure. This figure in itself gives an indication of how far beyond ordinary testing experience is the amount of energy released by compressed air. For reasons already mentioned, it does not represent fully the comparative destructive effect of compressed air. Thus, in the first place, the value of structural strain energy is over-estimated and, in the second place, no account is taken of the greater destructive effect of the energy released by compressed air, compared with the same amount of strain energy released in the structure.

Finally, the structural strain energy as calculated is seen (Fig. 3) to be some *two hundred* times that released by the water in a water pressure test. Although the total energy is small compared with that released in an air test, it merits a little consideration owing to a peculiarity of water testing often to be observed. Owing to the damping effect of water, failure is frequently local and a discharge of water takes place through the fracture. From the results obtained by these calculations it would appear that most of

the water discharge is attributable to contraction of the structure, rather than to expansion of the water.

2. ENERGY OF COMPRESSED AIR

2.1. Basic conditions

The available energy in compressed air is taken to be the work that would be done in resisting expansion down to a pressure equal to that of the outside atmosphere. For the present purpose there are two special conditions which have to be taken into consideration.

The first concerns the extent to which heat is transferred from the outside atmosphere. When a pressure cabin or other container bursts, the expansion takes place rapidly. It is reasonable, therefore, to assume that no heat is transferred and that expansion follows the adiabatic law. It is of interest to note that available energy calculated on this basis is the minimum, and that a greater amount of work would be done if the air, which cools during adiabatic expansion, received heat from the atmosphere.

The second condition is that the energy available for destructive purposes corresponds to work done by resultant forces, *i.e.* to the differential pressure in this case. In other words, there has to be deducted from the gross work done by the expanding air the work done against the static resistance of the atmosphere. This static resistance, it should be noted, has no connection with dynamic resistance associated with the inertia of the outside air, and would occur even if the density of air were zero while still exerting its usual pressure.

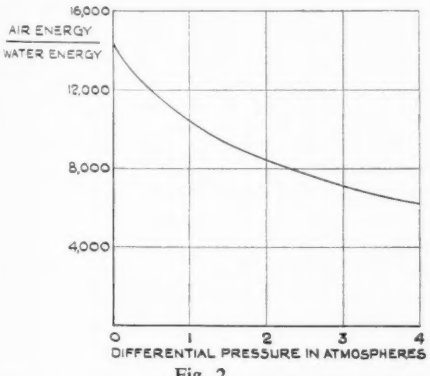


Fig. 2.
Comparison of destructive energy for air and water testing.

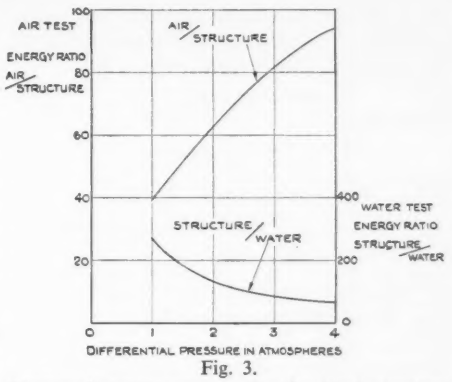


Fig. 3.
Structural strain energy ratios for ideal structure.

2.2. Mathematical formulæ

The foregoing principles may be expressed mathematically in a fairly simple form, with the following notation.

p = absolute pressure of the expanding air at any stage of the expansion
p_1 = the initial pressure
p_2 = the final pressure, *i.e.* the pressure of the external atmosphere
v = the volume of the expanding air at any stage of the expansion
v_1 = the initial volume of the air, *i.e.* the volume of the pressure cabin
v_2 = the final volume of the air

For the available energy (W) we have

$$W = W_1 - W_2 \quad . \quad . \quad (1)$$

where

$$W_1 = \int_1^2 p\,dv \quad . \quad . \quad (2)$$

and

$$W_2 = p_2 \int_1^2 dv \quad . \quad . \quad (3)$$

2.3. Adiabatic expansion

For adiabatic expansion we have the law,

$$pv^\gamma = C = \text{constant}, \quad . \quad . \quad (4)$$

and hence equation (2) becomes

$$W_1 = C \int_1^2 v^{-\gamma}\,dv$$

$$= \{ C/(\gamma - 1) \} \{ v_1^{1-\gamma} - v_2^{1-\gamma} \}.$$

Elimination of C and v_2 by means of equation (4) then leads to

$$W_1 = \frac{p_1 v_1}{(\gamma - 1)} \left\{ 1 - \left(\frac{p_1}{p_2} \right)^{\frac{1-\gamma}{\gamma}} \right\} \quad . \quad (5)$$

For the work done against the static resistance of the external atmosphere we have from equation (3)

$$W_2 = p_2 (v_2 - v_1)$$

which leads to

$$W_2 = p_2 v_1 \{ (p_1/p_2)^{1/\gamma} - 1 \} . \quad (6)$$

2.4. Non-dimensional forms

At this stage it is convenient to introduce a non-dimensional system and at the same time to express the initial internal pressure as a pressure difference between the inside and outside of the cabin.

The following notation is adopted for these purposes:—

$P = p_1 - p_2$ = initial difference of pressure between inside and outside

$Q = (p_1 - p_2)/p_2$ = initial pressure difference expressed as a multiple of the external pressure

$V = v_1$ = volume of the pressure cabin

Equation (5) may therefore be written

$$\frac{W_1}{PV} = \frac{p_1}{(p_1 - p_2)(\gamma - 1)} \left\{ 1 - \left(\frac{p_1}{p_2} \right)^{\frac{1-\gamma}{\gamma}} \right\}$$

Hence we may write for the non-dimensional value of the energy ($\overline{W}_1$),

$$\overline{W}_1 = \frac{W_1}{PV} = \frac{Q+1}{Q(\gamma - 1)} \left\{ 1 - (Q+1)^{\frac{1-\gamma}{\gamma}} \right\} \quad (7)$$

Similarly equation (6) may be written

$$\frac{W_2}{PV} = \frac{p_2}{p_1 - p_2} \left\{ \left(\frac{p_1}{p_2} \right)^{1/\gamma} - 1 \right\}$$

to give

$$\overline{W}_2 = W_2/(PV) = (1/Q) \{ (Q+1)^{1/\gamma} - 1 \} \quad (8)$$

For the net energy $\overline{W}$ expressed non-dimensionally we have

$$\overline{W} = W/(PV) = \overline{W}_1 - \overline{W}_2 . \quad (9)$$

2.5. Numerical values

For numerical computation we may take $\gamma = 1.4$ $(= 7/5)$ to obtain,

$$\overline{W}_1 = 2.5 \{ (Q+1)/Q \} \{ 1 - (Q+1)^{-2/7} \} \quad (10)$$

and

$$\overline{W}_2 = (1/Q) \{ (Q+1)^{5/7} - 1 \} . \quad (11)$$

Values for the energy coefficients are given in Table I.

The curves (Fig. 1) show the significance of the energy absorbed in overcoming the static resistance of the air at low values of Q, and generally within the normal working range of strength tests. If this deduction from the gross energy were to be ignored, therefore, radically different results would be obtained.

TABLE I

ENERGY COEFFICIENTS FOR COMPRESSED AIR

Q	$\overline{W}_1$	$\overline{W}_2$	$\overline{W}$
0	0.7143	0.7143	0
1	0.8984	0.6407	0.2577
2	1.0102	0.5959	0.4143
3	1.0902	0.5639	0.5263
4	1.1519	0.5392	0.6127
∞	2.5	0	2.5

In this table the values of the energy coefficients for Q tending to zero and to infinity respectively have been calculated as limits. The corresponding curves are plotted in Fig. 1.

The main concern, however, is with the curve for net energy. It is seen that the non-dimensional value for this starts from zero at zero value for Q and then increases rapidly. There is an upper limit to the value of the energy coefficient, namely 2.5, which is approached asymptotically.

2.7. Actual values of energy

Although the non-dimensional notation is the most useful for general comparative purposes, it is instructive to consider also the absolute values of energy released on failure in typical strength tests.

The average volume of eight pressure cabins that have been selected as representative of civil aircraft, is approximately 7,000 cubic feet and the average maximum working pressure may be taken as roughly half a ground level atmosphere. For a test at ground level with a factor of 3.0, therefore, the energy released may be calculated as follows:—

Differential pressure (Q) in standard atmospheres $= 3 \times 0.5 = 1.5$. The curves then give an energy coefficient

$$\overline{W} = 0.34.$$

Hence $W = 0.34 \times 1.5 \times 14.7 \times 144 \times 7{,}000$ ft./lb. where ground level atmospheric pressure is taken as 14.7 lb./sq. in.

Hence $W = 7.56 \times 10^6$ ft./lb.

$= 3.375$ ft./tons.

It is of interest that this amount of energy is roughly equal to the mechanical energy released by a 100 lb. bomb filled with T.N.T.

It does not necessarily follow from consideration of available energy alone that the same amount of damage would be done, but it is significant that such practical experience as there is, indicates that results are not dissimilar in cases where the cabin bursts into fragments.

The figure quoted for available energy of compressed air relates only to an average pressure cabin. A value some two or three times this could occur for a large aircraft designed for operation at high altitudes.

3. STRAIN ENERGY IN A PRESSURE CABIN STRUCTURE

3.1. Strain Energy of a Sphere

The simplest, and at the same time the most efficient, structure for holding compressed air is the sphere; and this is chosen for consideration first. The sphere is assumed to have a thin wall of uniform thickness, and to carry purely tensile stress. It is further assumed that the thickness is chosen to give a prescribed working stress at a given working pressure.

The following notation is used for this case:—

T = prescribed working stress
P = corresponding differential pressure
V = volume
D = diameter
t = thickness
E = Young's modulus
m = Poisson's ratio

Taking a plane section through a diameter we have for equilibrium of a hemisphere:—

$$T = DP/(4t) \quad . \quad . \quad (12)$$

For elastic distortion, the linear strain, which is the same in all directions, is therefore:—

$$(T/E)(1-m)$$

The volumetric strain is therefore

$$dV/V = (3T/E)(1-m)$$

Hence since strain energy (W) is given by

$$W = \tfrac{1}{2} P\,dV$$

the strain energy coefficient $\overline{W}$ becomes

$$\overline{W} = W/(PV) = \tfrac{3}{2}(1-m)\,T/E. \quad (14)$$

3.2. Simple Cylinder

The next case is the simple cylinder in which end effects are virtually ignored. Thus the ends may be regarded as flat plates that are infinitely stiff in bending but impose no radial constraint on the curved walls of the cylinder.

The following further notation is introduced for the cylinder:—

T = hoop stress
S = longitudinal stress
l = length
c = circumference;

where T and S are clearly principal stresses.

By taking a plane section through the axis of the cylinder we have for equilibrium

$$T = DP/(2t). \quad . \quad . \quad (15)$$

By taking a plane section perpendicular to the axis we have for equilibrium

$$S = DP/(4t) = T/2. \quad . \quad (16)$$

It is to be noted that the hoop stress (T) is twice the longitudinal stress (S). The hoop stress (T) is here taken as the critical stress for design.

The hoop strain is given by

$$\frac{dc}{c} = \frac{T - mS}{E} = \frac{(2-m)}{2}\frac{T}{E} \quad . \quad (17)$$

The longitudinal strain is given by

$$\frac{dl}{l} = \frac{S - mT}{E} = \frac{(1-2m)}{2}\frac{T}{E} \quad (18)$$

Hence the volumetric strain is given by

$$\frac{dV}{V} = \frac{2dc}{c} + \frac{dl}{l} \quad . \quad . \quad (19)$$

Since strain energy W is given by

$$W = \tfrac{1}{2} P dV,$$

we get the strain energy coefficient ($\overline{W}$),

$$\overline{W} = \frac{W}{PV} = \frac{(5-4m)}{4}\frac{T}{E} \quad . \quad (20)$$

3.3. Cylinder with Domed Ends

The next case for consideration is a cylinder with hemispherical ends, which is readily treated by combination of the two previous cases. It is assumed that the respective thicknesses of sphere and cylinder are adjusted to give the same maximum principal stress (T). In other words, the stress in the two hemispheres is equal to the hoop stress in the cylinder, a condition which requires the hemispheres to have half the thickness of the cylinder. This does not quite give equal strains at the junctions owing to different Poisson's ratio effects, but it is considered that the discrepancy can be ignored in a rough general analysis.

The energy stored in the domed cylinder is obtained by appropriately combining the results already obtained for the sphere and the simple cylinder separately. This leads

to a strain energy coefficient given by

$$\overline{W}=\frac{W}{PV}=\tfrac{3}{4}\left\{\frac{4(1-m)+(5-4m)\lambda}{(2+3\lambda)}\right\}\frac{T}{E} \quad (21)$$

where $\lambda=l/D$.

3.4. Numerical values

It is now proposed to take $m=0.3$ as appropriate for light alloy generally. This gives the energy coefficient

$$\overline{W}=\frac{W}{PV}=\frac{3}{2}\frac{(1.4+1.9\lambda)}{(2+3\lambda)} \quad . \quad (22)$$

For the sphere we then have

$$\overline{W}=1.05\,T/E.$$

For the simple cylinder we have

$$\overline{W}=0.95\,T/E.$$

The corresponding values for the cylinder with domed ends depends upon the length/diameter ratio (λ). Thus,

$$\text{for } \lambda=1,\quad \overline{W}=0.99\,T/E$$
$$\text{for } \lambda=2,\quad \overline{W}=0.97\,T/E.$$

The interesting feature of this result is the limited range of variation of strain energy coefficient. Thus with 1.05 and 0.95 as extreme values we may write

$$\overline{W}=W/(PV)=T/\mathrm{E} \quad . \quad (23)$$

to obtain a formula applicable with fair accuracy to all reasonable combinations of sphere and cylinder.

3.5. Practical application

Before these results can be applied to real structures two limitations of the analytical work have to be considered, namely the idealistic character of the structures assumed and the assumption that distortion is elastic.

It is clear that elastic distortion does not apply to a structure loaded to destruction. We are concerned, however, not with the energy absorbed by the structure but with the energy that can be released. Energy absorbed in plastic deformation, therefore, can be ignored. It is thus considered reasonable to take the ultimate tensile stress of the material for T in the formula in association with the value of E corresponding to the elastic range.

As representative of good light alloy the following numerical values are taken

$$T=65{,}000 \text{ lb./sq. in.}$$
$$E=10{,}000{,}000 \text{ lb./sq. in.}$$

These values then give an energy coefficient from equation (23) of

$$\overline{W}=W/(PV)=6.5\times10^{-3} \quad . \quad (24)$$

It is clear that a real pressure cabin structure cannot be strictly represented by the ideal system assumed. For this reason the results are only to be regarded as giving a rough indication of the strain energy in an actual pressure cabin. It is significant, however, that departure from the ideal will usually lead to lower average stresses. In consequence the strain energy of an actual pressure cabin will be somewhat less than that obtained for the ideal system.

4. Comparison of Stored Energy with That for Water

4.1. Energy for water

There has now to be considered the effect of using water instead of air for testing a pressure cabin.

The same notation is adopted as for air pressure, the principal parameters being reproduced for convenience.

V = volume of pressure cabin

P = differential pressure

Q = differential pressure expressed as a multiple of external pressure.

Since the results are to be applied solely to tests at ground level, the quantity Q becomes simply the differential pressure in atmospheres.

For water we have a compressibility C given by Kaye and Laby as 49.5×10^{-6} per atmosphere.

Hence we write for ground level conditions

$$dV/V=CQ,$$

and since the energy W is

$$W=\tfrac{1}{2}PdV,$$

the energy coefficient $\overline{W}$ is given by

$$\overline{W}=W/(PV)=\tfrac{1}{2}CQ, \quad (25)$$

TABLE II

ENERGY COEFFICIENTS

Differential Pressure in Atmospheres P	Value of Energy Coefficient $W/(PV)$		
	Ideal Structure	Air	Water
0	—	0	0
1	6.5×10^{-3}	0.2577	24.75×10^{-6}
2	6.5×10^{-3}	0.4143	49.50×10^{-6}
3	6.5×10^{-3}	0.5263	74.25×10^{-6}
4	6.5×10^{-3}	0.6127	99.00×10^{-6}

or taking the numerical value quoted above for C we have

$$\overline{W} = W/(PV) = 24.75\, Q \times 10^{-6}. \quad (26)$$

4.2. Comparative figures

It is now possible to draw up Table II for the available energy stored in air and water, and also in the idealised structure.

In this table a range of pressures is considered up to four atmospheres, which figure is not likely to be exceeded. The values for air and water corresponding to zero differential pressure (Q) have been calculated as limits.

It is important to bear in mind that the figures for structural strain energy apply only for cases where the structure reaches its maximum stress at the stipulated pressure. The particular values tabulated for different pressures therefore, correspond to different structures. Variation of strain energy coefficient with pressure for a *given structure* obeys a linear relationship between energy coefficient and pressure.

Figures for strain energy become unrealistic at low working pressures, since they refer to impracticably thin shells.

4.3. Energy ratios

Table III gives figures for the more interesting ratios between the various forms of stored energy for quick reference. These results are plotted in Figs. 2 and 3 and lead to the conclusions outlined in the Introduction.

5. ACKNOWLEDGMENT

Acknowledgment is made by the author to the Chief Scientist, Ministry of Supply, for permission to publish this paper.

Table III

Differential Pressure in Atmospheres	Energy Ratios		
	Air/Water	Air/Structure	Structure/Water
0 (limit)	14,430	—	—
1	10,410	39.6	26.2
2	8,370	63.7	13.1
3	7,090	81.0	87.5
4	6,190	94.3	65.7

ELECTRIC RESISTANCE WELDING AND ITS USE IN THE CONSTRUCTION OF AIRCRAFT

by

E. C. SYKES

INTRODUCTION

Designers and technicians pay a great deal of attention to the development of jointing methods for use in the construction of aircraft. A satisfactory method should be neat, economical in labour, flexible in application, and capable of yielding consistent results with a low weight penalty.

Electrical resistance welding meets the above requirements and it is with this process that I shall deal.

USES

Spot welding and seam welding are the two methods of resistance welding most used in aircraft construction, and they may be used to effect a joint between materials of a not too dissimilar nature and of which one element is in sheet form. It is possible and practicable to weld together different gauges, or more than two thicknesses of sheet.

The processes are largely used for the building up of big or awkwardly shaped components, and for the attachment of stiffening members. Seam welding provides a quick method of making pressure-tight joints, without recourse to the use of expensive and troublesome gaskets or sealing media.

DESIGN

Seam welding is merely spot welding in which the spots overlap, and since spot welding, for most purposes, may be regarded as "riveting without rivets," no new problems are raised in the design of the joint. The design, however, must be such that there is the necessary clearance for electrodes, and that the components can be satisfactorily cleaned. The welding may be placed in tension, but its strength is by far the greater in shear. In general, the shear strength of spot welds should be at least of the order of $f = T \times UTS \times K$

where

f in lb.

T = thickness of material in inches

Ultimate Tensile Stress in tons/in.^{2}

$K = 500$ for aluminium

$K = 750$ for steels.

It should be borne in mind that a spot welded joint is considerably more rigid than its riveted counterpart, and therefore will admit of less deformation before failure occurs.

Resistance welding is much lighter, cheaper and more resistant to fatigue and corrosion than riveting and, in addition, a perfectly smooth exterior surface can be obtained without costly countersinking and finishing operations.

THE PROCESS

Briefly, resistance welding is effected by passing a heavy electrical current through the pieces of metal to be joined. The current heats the metal to welding temperature in exactly the same way as it heats the element of an electric fire, and the parts are then forced together by pressure.

Now let us examine the heating process more closely. When an electrical current passes through a resistive conductor, heat is developed according to the formula: $H = ri^2t$ (r = resistance i = current t = time). It will be seen that the amount of heat developed is dependent on a number of factors. Considering the time factor first, it is evident that much of the heat will be

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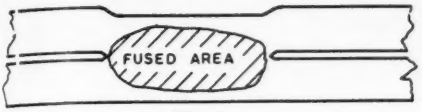


Fig. 1.
Section through spot weld.

conducted away and wasted unless the metal is raised to welding temperature within a relatively short space of time. This time should also be as short as possible to increase the speed of production. Times used in practice vary with the type of machine, but are of the order of 1/100th second for thin gauges, or up to 2 seconds for ½ in. steel plate. The time during which the current flows may be under the control of the operator, but is usually controlled by pre-set electro-mechanical mechanisms.

The second factor controlling the amount of heat developed is the resistance of the joint to the flow of current. This may be divided into four portions: the resistance of the metal itself, the resistance of the very thin superficial film on the metal, the resistance of the contact between the pieces being joined, and lastly, the resistance of the contact between the work and the electrodes carrying the welding current.

(*a*) The resistance of the metal is very low and may be neglected for all practical considerations, being but 0.05 $\mu\Omega$/in.^{2} for 20 G Aluminium

(*b*) The resistance of the surface film is low, being only perhaps 0.7 $\mu\Omega$ per sq. in. for properly cleaned aluminium. Nevertheless, this resistance is the principal active resistance; there are, of course, two such resistances at the faying surface of the joint.

(*c*) The resistance of the contact between the work pieces will vary with the pressure holding the pieces together. Ideally, this resistance should be fairly high, but in order to obtain a sufficiently uniform value, it has been found desirable to apply a pressure sometimes as high as 15,000 lb./sq. in. This high pressure greatly reduces the resistance of the contact, lowering it to a value of about 1.0 $\mu\Omega$ or less per sq. in. in the case of cleaned aluminium. In some welding machines for aluminium, the pressure is almost halved just as the weld current begins to flow; this raises the resistance considerably.

(*d*) The resistance of the contact between the current conductors and the work is kept as small as possible and practically all the heat developed is carried away by the water-cooled conductors.

The two factors already examined are more or less fixed, and thus it follows that control of the amount of heat developed must be performed largely by varying the third factor, *i.e.* the current. The time and resistance being so small, the welding current must be large, in order to develop a sufficient amount of heat. For welding aluminium this current may exceed 2,000,000 amps./sq. in. of weld area for the thicker gauges, while for steel the amperage required is lower, being about 400,000 amps./sq. in. The thicker the gauge of the material, the larger the weld required, and the larger the current.

There is another factor affecting the welding, although in an entirely different way, and this is the pressure applied after the metal has been heated. This pressure "forges" the weld and then "follows up" the contraction as the weld cools, thus preventing the formation of internal cracks and cavities. In some machines, when welding aluminium, the pressure is increased considerably after the weld current has passed, in order to enhance the forging effect.

Having examined the generation of the necessary heat, the welding machine itself must now be discussed, and then methods of preparing the work for welding. The machine may be considered as consisting of four units which may, or may not, be combined to form one assembly. These sections are (*a*) the mechanism for generating and controlling the heavy welding current, (*b*) the conductors to carry the welding current to the work, (*c*) the mechanism for controlling and applying pressure to the joint, and

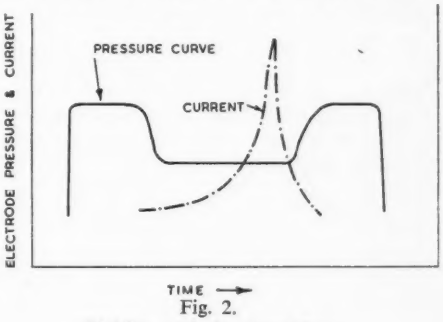


Fig. 2.
Welding cycle for aluminium.

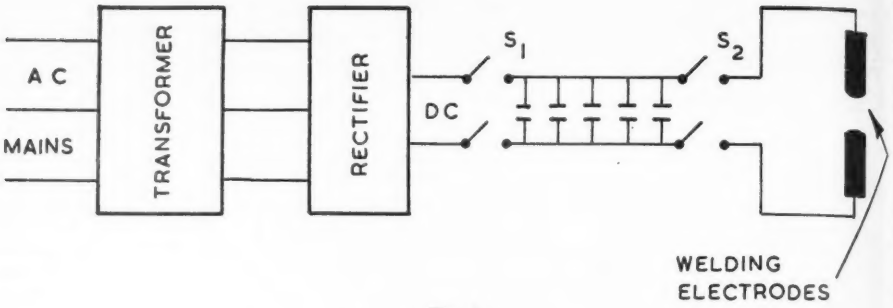


Fig. 3.
(*a*) Battery or condenser type.

Operation: Batteries (or condensers) are charged continuously with S_1 closed and S_2 open. S_1 is then opened and S_2 closed, when a heavy current passes through the welding electrodes.

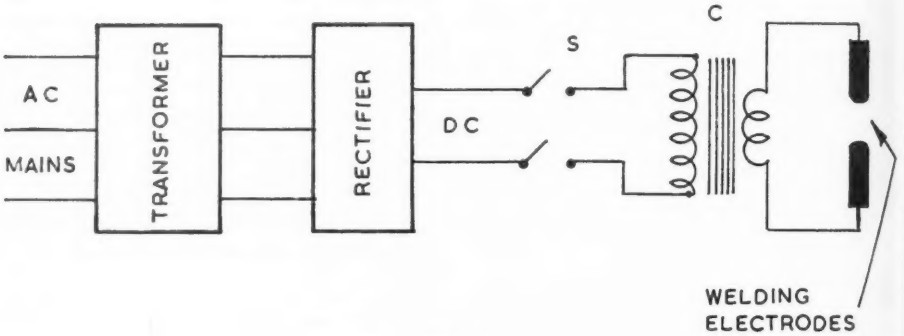


Fig. 3.
(*b*) Choke type.

Operation: The switch S is closed and a strong D.C. current flows through the winding of the choke C and produces a very powerful magnetic field. When the switch is reopened, this magnetic field collapses very rapidly, and a surge is induced in the electrode circuit.

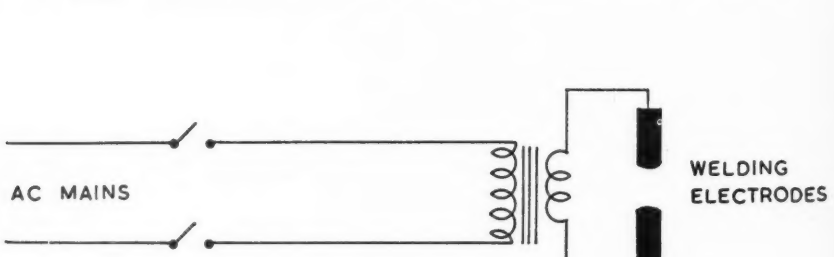


Fig. 3.
(*c*) Transformer type.

(*d*) a timing system to co-ordinate properly the pressure and current Let us consider these one by one.

(*a*) There are three ways commonly used for producing the heavy welding current, namely:

(1) The storing up of energy in a battery of continuously charged secondary cells (*i.e.* accumulators) and drawing directly from these.

(2) The accumulation of electrical energy in a choke or a bank of large electrolytic condensers, and then permitting rapid discharge.

(3) The third method is to transform an alternating current to produce a high amperage at low voltage.

All three methods raise different supply problems. The accumulator storage is perhaps the easiest since the demand is relatively small and continuous. The choke or condenser type of machine requires perhaps 350 amps, intermittently for a second or so, but this load is balanced across the mains, whereas the transformer type takes an unbalanced load. For this reason this latter method is used mainly for the smaller type of machine. To prevent over-heating of the power unit, all such units are fan-cooled. Choke-type machines are usually enclosed in a heavy steel casing which acts as a shield to protect surrounding instruments from the extremely powerful magnetic field produced when the machine is functioning.

(*b*) The conductors are usually made of heavy section high conductivity copper bar or braid, and are often water-cooled. For spot welding, they terminate in replaceable tips or electrodes of hard drawn copper, or copper alloy, which are tapered to concentrate the current at the location of the weld, while for seam welding, the electrodes consist of two copper alloy wheels, one of which is power-driven.

(*c*) Pressure is applied to the electrodes in the smaller machines by a hand- or foot-operated lever system, while for medium-sized machines part of the pressure is obtained from a large solenoid acting through the same foot-operated lever system. The high pressure required for large welding machines is almost universally applied by pneumatic cylinders, acting either directly on the electrodes, or through a simple rocker lever system.

The pressure system may be built on to the casing of the power unit, or may be mobile in the sense that it is suspended from an overhead tramway.

(*d*) The time for which the current passes is controlled by electrical or mechanical means of various types. This controller is often inter-linked with the mechanism governing the application and variations of the pressure, and if the machine is fully automatic (as seam welders always are) the controller also links up with the speed regulator. This inter-linking ensures identical welding cycles of the correct sequence and timing.

GENERAL

A complete welding cycle takes from one to three seconds, but multi-head machines have been built which greatly increase the production rate. In spite of the tremendous currents there is no danger of electrocution of the operator because of the low voltage and the virtual short-circuiting through the work.

A complete installation for welding aluminium aircraft components requires about 50 sq. ft. of space and weighs about 9 tons. There being no vibration, deep machine foundations are unnecessary.

Before leaving the welding machine there is one point, often overlooked, which can make all the difference between a good or bad weld. It is absolutely essential that the welding machine be kept clean and maintained in perfect electrical and mechanical order.

The electrodes are a part of the electrical system and therefore they must be kept free from pick up and maintained at the correct tip size. This is done by filing and smoothing with fine emery paper, with an occasional re-machining to shape when badly worn.

CLEANING

Before material can be welded, it must be brought to a satisfactory surface finish. Grease, oil, paint, and so on, are insulators and must be removed. Scale and oxide on ferrous materials, while not insulators, have a very high resistance and must also be removed; this is usually done by immersion in a hydrochloric acid solution, by scratch brushing or by shot blasting. Ferrous materials then need no further preparation before welding.

Aluminium alloys, after degreasing, are generally prepared for production purposes by etching with acids (usually hydrofluoric or phosphoric) in order to remove the thin adherent oxide coating. This etching may be performed either by immersion or by the application of the acid compounded with other chemicals to form a paste. The acid is removed by vigorous washing with water. A pickle smut forms on Dural-type alloys and this is removed by immersion in nitric acid, followed by washing with water.

Another method is treatment with a mild abrasive such as metal polish, but this process is laborious and rather uneconomical for large-scale production work.

Another method, much used in the United States, is to scratch brush the metal lightly, but this is a somewhat delicate operation for thin gauge alclad materials and the surface produced is inclined to be affected by exposure to air.

Magnesium alloys may be cleaned by similar methods.

All cleaning must be done meticulously. Failure to do so will result in over-heated welds in steel, while in light alloys dirt causes such a rapid development of heat that the metal is fused with explosive violence for about half an inch around the weld. This occurrence is commonly known as a " blow out " and the molten metal may cause serious injury unless the operator is wearing protective clothing and goggles.

INSPECTION AND PROCESS CONTROL

Welds are not easy things to inspect, as visual examination reveals only surface defects such as cracks or severe over-heating. In order to keep a check on the strength of the welds being produced, therefore, it is usual to demand daily samples from each machine and to determine the shear strength of the spots by destructive testing. A shop method, which provides a good, but not quantitative result, is to wel[1] together two strips of metal and then to " peel " them apart. Good welds will tear through the sheet and will exhibit uniform cylindrical weld plugs.

Occasionally it is desired to check that the penetration of the welding is satisfactory, and this is done by sectioning through a weld and polishing and etching to reveal the macro-structure of the joint.

A usual figure for acceptable penetration on aluminium alloys is 70 per cent. This macro examination also reveals internal cracks and cavities. In addition to these checks it is good practice to test each spot weld individually. For normal gauge light alloy components, it is usually sufficient to insert a thin knife blade under the edge of the joint and give it a slight twist, whereupon a poor spot breaks with a clearly audible crack.

Steel welds cannot be tested in this way, but defective spots may be shown up by a variation of the technique used in electro-magnetic crack detection.

The device consists of a thin transparent plastic cell with a white translucent bottom and containing a suspension of fine iron oxide in paraffin. A strong magnet is placed on one side of the work and the cell on the other. If the weld has been properly made the magnetic characteristics of the metal of the weld slug are altered, and this is indicated as a white spot. If the weld is unsatisfactory, no white spot appears in the fluid.

If desired, it would be possible to submit the welding to X-ray examination. This could be done conveniently by viewing on a fluorescent screen.

SUMMARY

Summing up, it can be said that resistance welding is a sound and proven method of making joints in aircraft structures. With intelligent design and operating supervision a joint, far neater and stronger than its riveted counterpart may be produced more cheaply and much more quickly.

ANALYTICAL TREATMENT OF LINKED LEVERS AND ALLIED MECHANISMS

by

B. C. CARTER A.R.C.Sc., D.I.C., F.R.Ae.S.

1. INTRODUCTION

In systems for transmitting control motion there may be two shafts set in any relative position, each carrying a lever arm, and the ends of the lever arms may be connected by a link having ball ends. In general, the range of working movement for such a system is somewhat limited and the relationship between the angular movements of the two shafts is non-linear.

Although for some purposes it may suffice to mock-up the system and determine the motion experimentally, for other purposes analytical treatment may be required. The general solution given below is readily applied to particular cases, which may be simpler.

2. CO-ORDINATE GEOMETRY OF THE SYSTEM

Referring to Fig. 1, axis OX is taken along one shaft, OP, and axis OZ is taken along OR, the common perpendicular to the shafts, which has length c. The other shaft, RS, lies in a plane parallel to XOY and its direction is defined by the angle α it makes in this plane with the parallel to OP, measured in the positive sense.

The end Q of the lever on OP moves in a circular arc about P in a plane parallel to YOZ, and the position of Q is defined by the length $PQ=A$ and the angle θ, together with the length $OP=a$.

Again, the end T of the lever on RS moves in a plane normal to the shaft through T, and the position of T is defined by the length $ST=B$ and the angle α, together with the length $RS=b$ and the angle ϕ measured in the positive sense from the line through S parallel to OZ. It will be noted that θ also is measured from a line parallel to OZ (through P), in the positive sense.

Paper received October 1949.

The co-ordinates of Q are $x_1\ y_1\ z_1$ where:

$$x_1=a;\ y_1=-A\sin\theta;\ z_1=A\cos\theta \quad . \quad (1)$$

If d denote the length of the link QT, the point T must lie on the surface of a sphere of radius d having its centre at Q. The equation to this sphere is clearly:

$$(x-a)^2+(y+A\sin\theta)^2+(z-A\cos\theta)^2=d^2 \quad . \quad . \quad . \quad . \quad (2)$$

The co-ordinates of T are seen by reference to Fig. 2 to be $x_2\ y_2\ z_2$ where:

$$x_2=b\cos\alpha+B\sin\phi\sin\alpha \quad . \quad (3)$$

$$y_2=b\sin\alpha-B\sin\phi\cos\alpha \quad . \quad (4)$$

$$z_2=c+B\cos\phi \quad . \quad . \quad . \quad (5)$$

By substituting these values, respectively, for $x\ y\ z$ in (2), the following general solution is obtained, connecting all the quantities concerned and giving d explicitly in terms of the remaining quantities:

$$(b\cos\alpha+B\sin\phi\sin\alpha-a)^2+ +(b\sin\alpha-B\sin\phi\cos\alpha+A\sin\theta)^2+ +(c+B\cos\phi-A\cos\theta)^2-d^2=0 \quad . \quad . \quad . \quad . \quad (6)$$

Equation (6) can be transformed into

$$H-J\sin\theta=K\cos\theta$$

or

$$\sin\theta=HJ\pm K\sqrt{(J^2+K^2-H^2)}/(J^2+K^2), \quad . \quad . \quad . \quad . \quad (6a)$$

where $H=a^2+b^2+c^2+A^2+B^2-d^2-2\,ab\cos\alpha-2\,aB\sin\phi\sin\alpha+2\,Bc\cos\phi$

$$J=2\,A\,(B\sin\phi\cos\alpha-b\sin\alpha)$$

$$K=2\,A\,(B\cos\phi+c).$$

The value of $\sin\theta$ must come, of course, within the range plus and minus one and it is unreal when $H^2>J^2+K^2$.

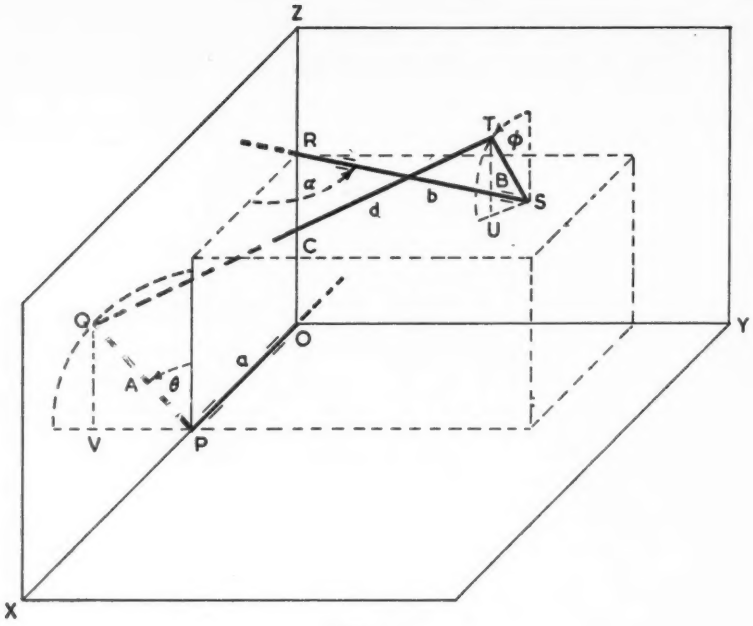


Fig. 1.

Again, equation (6) can be transformed into

$$U - V \sin\phi = W \cos\phi \quad \text{or}$$

$$\sin\phi = [UV + W\sqrt{(V^2 + W^2 - U^2)}]/(V^2 + W^2), \qquad (6b)$$

where $U = a^2 + b^2 + c^2 + A^2 + B^2 - d^2 - 2ab\cos\alpha + 2Ab\sin\theta\sin\alpha - 2Ac\cos\theta$

$$V = 2B(A\sin\theta\cos\alpha + a\sin\alpha)$$

$$W = 2B(A\cos\theta - c).$$

These derive from HJK by putting θ for ϕ; b for a; a for b; $-B$ for A; $-A$ for B. The changes of sign are an outcome of the sign conventions.

The value of $\sin\phi$ must come, of course, within the range plus and minus one and it is unreal when $U^2 > V^2 + W^2$.

Thus θ is obtained in terms of ϕ from (6*a*) and ϕ in terms of θ from (6*b*), as may be required, and the whole behaviour of the mechanism may be studied precisely.

Equation (6) may be used for investigating other mechanisms by treating different quantities as variables. Suppose, for example, ST is a lever of a valve mechanism and Q is a tappet constrained to move parallel to OX. Then A and θ serve to define the position of the tappet in relation to the remainder of the mechanism, and a may be treated as the prime variable in the motion. Furthermore, if there are changes in basic distances due to thermal expansion or contraction, the effects of such changes can be assessed by using (6).

It will be found that for special cases (as when the shafts intersect at right angles, for example) equation (6) reduces to a much simpler form such as may lend itself to study of velocity and acceleration derivatives of variables of interest.

In using the analysis, the physical significance of results should be noted so that unsatisfactory settings shall be avoided.

The quantities a, b, c, θ and ϕ can have positive or negative values individually; there is less liability to mistake if A and B are kept positive and values are taken accordingly for θ and ϕ. The angle α and the length d may as well be kept positive.

3. FORCES AND TORQUES

To investigate forces and torques, the procedure is as follows: The direction

cosines of link QT are l, m, n, where:

$$l=(x_2-x_1)/d;\ m=(y_2-y_1)/d;$$
$$n=(z_2-z_1)/d. \quad . \quad . \quad (7)$$

If F be the force at Q that the lever PQ exerts along QT, this force has components lF, mF, and nF parallel to OX, OY and OZ, respectively. Thus if the torque operating in shaft OP be denoted by Q_a,

$$Q_a=-mFA\cos\theta-nFA\sin\theta \quad . \quad (8)$$

The component forces acting at T on the lever ST are the same as those acting on the link QT at Q, and they produce a turning moment Q_b about shaft RS, where

$$Q_b=(lF\ \sin\alpha-mF\cos\alpha)\,B\cos\phi-nFB\sin\phi \quad . \quad . \quad . \quad . \quad (9)$$

By the principle of virtual work, when θ and ϕ are the variables:

$$Q_a\,d\theta=Q_b\,d\phi \quad . \quad . \quad (10)$$

Thus $\dfrac{d\theta}{d\phi}=\dfrac{Q_b}{Q_a}$

$$=-\frac{B}{A}\frac{(l\sin\alpha-m\cos\alpha)\cos\phi-n\sin\phi}{m\cos\theta+n\sin\theta} \quad . \quad . \quad . \quad . \quad (11)$$

This result may be checked by differentiating (6) with regard to ϕ.

At dead centres, both the numerator and the denominator of the expression for $d\theta/d\phi$ given in (11) become zero, whence

$$l\sin\alpha+n(\cos\alpha\tan\theta-\tan\phi)=0$$

i.e. $(b\cos\alpha+B\sin\phi\sin\alpha-a)\sin\alpha+$
$+(c+B\cos\phi-A\cos\theta)\times$
$\times(\cos\alpha\tan\theta-\tan\phi)=0,$

which may be used as a criterion for dead centre conditions.

The forces acting along OX, OY, and OZ are lF, mF, and nF, respectively, and may be evaluated accordingly.

The reaction force $-F$ on the lever end at Q may be replaced by a parallel force $-F$ at O plus moments X_1, Y_1, Z_1 about OX, OY, and OZ, respectively where:

$$X_1=-nFy_1+mFz_1 \quad . \quad (12)$$

$$Y_1=-lFz_1+nFx_1 \quad . \quad . \quad (13)$$

$$Z_1=-mFx_1+lFy_1 \quad . \quad . \quad (14)$$

Here, $X_1=-Q_a$ and (12) accords with (8).

The reaction couples Y_1 and Z_1 come on to the bearings of shaft OP and it is of interest to examine them further.

The axis of the resultant of these two couples is normal to OX and makes an angle γ with OY, where:

$$\tan\gamma=(-mx_1+ly_1)/(-lz_1+nx_1) \quad . \quad (15)$$

If R_1 denote the resultant of the two couples,

$$R_1^2=Y_1^2+Z_1^2$$
$$=F^2\,[l^2z_1^2+(n^2+m^2)x_1^2+l^2y_1^2-$$
$$-2lnz_1x_1-2lmx_1y_1]$$
$$=F^2\,[l^2(y_1^2+z_1^2-x_1^2)+x_1^2-$$
$$-2lx_1(nz_1+my_1)]$$

$$R_1=F\,[l^2(A^2-a^2)+a^2-$$
$$-2Aal(n\cos\theta-m\sin\theta)]^{\frac{1}{2}} \quad . \quad . \quad . \quad . \quad (16)$$

$$Q_a=F(An\sin\theta+Am\cos\theta)$$
$$R_1/Q_a=[l^2(A^2-a^2)+a^2+2Aal(m\sin\theta-$$
$$-n\cos\theta)]^{\frac{1}{2}}/[Am\cos\theta+An\sin\theta] \quad . \quad . \quad . \quad . \quad (17)$$

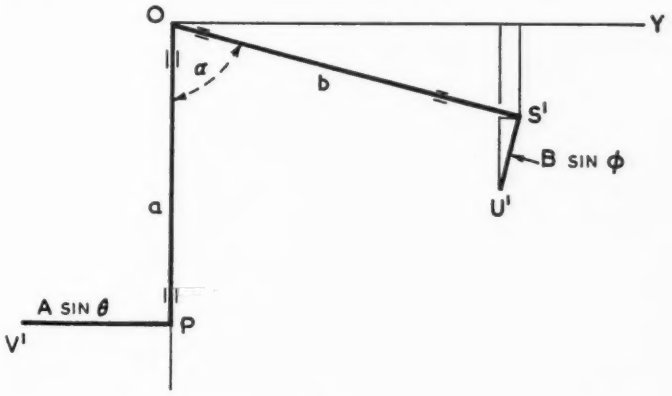


Fig. 2.

Fig. 3.
P, *S* and *R* coincide with origin *O*.
Hooke's universal joint.

Fig. 4.
Cardan shaft system with Hooke's joints.

Thus both F and R_1 can be found in terms of Q_a.

The force F on the lever end at T may be replaced by a parallel force at O (neutralising the previous force) plus turning moments X_2, Y_2, and Z_2 about OX, OY, and OZ, respectively, where,

$$X_2 = nFy_2 - mFz_2 \quad . \quad . \quad (18)$$

$$Y_2 = lFz_2 - nFx_2 \quad . \quad . \quad (19)$$

$$Z_2 = mFx_2 - lFy_2 \quad . \quad . \quad (20)$$

From these equations, the forces on the shaft RS and its bearings may be derived; the torque Q_b applied to shaft RS has been given in terms of driving torque Q_a in (11).

4. SPECIAL CASE OF HOOKE'S UNIVERSAL JOINT

Whereas the general case leads to some rather complex expressions, considerable simplification occurs in some special cases, notably, in the case of the Hooke's joint (*see* Fig. 3). This joint has an action which would be unchanged, when the shafts are located endwise, if the cruciform spider were omitted and a ball-ended link were introduced connecting a fork-end of one shaft with a fork-end of the other shaft. The other two fork-ends may likewise be connected and, if the two links be secured together parallel, there is no need to have the shafts located endwise for the location will be fixed by the coupling arrangement.

Instead of a link or two links, four cords may be used connecting the fork-ends in a parallelogram and making it possible to drive in either sense of rotation. Thus it can be seen how a Hooke's joint transforms into a flexible disc coupling.

We have thus a special case in which a, b, and c, are all zero and, as the spider is right-angled, it follows that

$$A^2 + B^2 = d^2 \quad . \quad . \quad . \quad (21)$$

Equations (3), (4) and (5) become

$$x_2 = B \sin\phi \sin\alpha \quad . \quad (3')$$

$$y_2 = -B \sin\phi \cos\alpha \quad (4')$$

$$z_2 = B \cos\phi \quad . \quad . \quad (5')$$

Thus (6) becomes

$$B^2 \sin^2\phi \sin^2\alpha + (-B \sin\phi \cos\alpha + A \sin\theta)^2 + \\ + (B \cos\phi - A \cos\theta)^2 - d^2 = 0$$

By using (21), this reduces to

$$\tan\theta \tan\phi \cos\alpha + 1 = 0 \quad . \quad (6')$$

Differentiating,

$$\sec^2\theta \frac{d\theta}{d\phi} \tan\phi \cos\alpha + \sec^2\phi \tan\theta \cos\alpha = 0$$

. . . . . (22)

By using (6′) for eliminating ϕ,

$$\frac{Q_b}{Q_a} = \frac{d\theta}{d\phi} = -\frac{\sin 2\theta}{\sin 2\phi} = \frac{1 - \sin^2\theta \sin^2\alpha}{\cos\alpha}$$

. . . . . (23)

where Q_b and Q_a relate to torques corresponding to the one link conveying force F. Equation (15) becomes

$$\tan\gamma = -\frac{y_1}{z_1} = \tan\theta \quad . \quad . (15')$$

Thus the resultant reaction couple R_1 acts in the plane of the fork of shaft OX.
By using (6′), equation (17) reduces to

$$R_1/Q_a = l/(m\cos\theta + n\sin\theta)$$
$$= B\sin\phi\sin\alpha / \{(-B\sin\phi\cos\alpha + \\ + A\sin\theta)\cos\theta + (B\cos\phi - \\ - A\cos\theta)\sin\theta\}$$
$$= -\tan\alpha\cos\theta \quad . \quad . \quad . \quad (17')$$

The reaction couple $-Q_a \tan\alpha \cos\theta$ which rotates with the shaft thus varies harmonically. This gives particular interest to the fixed reaction couples Y_1 and Z_1.

From (13),

$$Y_1 = -lFz_1 \quad . \quad . \quad (13')$$

From (17),

$$Q_a = FA(n\sin\theta + m\cos\theta)$$

Thus

$$Y_1/Q_a = [l/(m\cos\theta + n\sin\theta)]\,[-z_1/A]$$
$$= (R_1/Q_a)(-\cos\theta)$$

Therefore

$$Y_1 = Q_a \tan\alpha \cos^2\theta$$
$$= Q_a(\tfrac{1}{2}\tan\alpha)(1 + \cos 2\theta) \quad . \quad (24)$$

which shows that Y_1 does not change sign and that it varies harmonically at twice the frequency of shaft rotation.

From (14),

$$Z_1 = lFy_1$$

Thus

$$Z_1/Q_a = [l/(m\cos\theta + n\sin\theta)]\,[y_1/A]$$
$$= (R_1/Q_a)(-\sin\theta)$$

Therefore

$$Z_1 = Q_a (\tfrac{1}{2} \tan \alpha)(\sin 2\theta) \quad . \quad . \quad (25)$$

Thus Z_1 varies harmonically at twice the frequency of shaft rotation and changes sign. The range of variation is the same as for Y_1 but the maximum value of Y_1 is twice the maximum value of Z_1*.

The force acting along shaft OP is $-lF$ and from (16) and (17′) it follows that this force is $(Q_a \tan \alpha \cos \theta)/A$. Thus, if α were in the region of $\pi/2$, the end force would be very large except for values of $\cos \theta$ near to zero; practical values for α must lie in the region of π or zero. The latter relates to one shaft being tubular and the other shaft running through it.

The force acting along the driven shaft may be derived similarly, as may the forces on the spider and its bearings in the forks.

5. APPLICATION TO A CARDAN SHAFT WITH HOOKE'S JOINTS

If a Cardan shaft (*see* Fig. 4) has a Hooke's joint at each end (with the forks on the shaft correctly set in the same plane) connecting to a further shaft in the same plane at angle β to the Cardan shaft, with the fork on this further shaft at angle ψ to OZ, we may write, on the basis of (6′)

$$\tan \phi \tan \psi \cos \beta + 1 = 0 \quad . \quad (6'')$$

From (6′) and (6″)

$$\tan \theta \cos \alpha = \tan \psi \cos \beta \quad . \quad (26)$$

Thus if $\beta = \pm \alpha$, then $\theta = \psi$ or $\psi + \pi$ and $d\theta/d\psi = 1$, so that the primary and final shafts have unit velocity ratio in such a case.

* Attention to the existence and character of the foregoing reaction couples is drawn by H. I. F. Evernden in a paper contributed recently to the Institution of Mechanical Engineers (Automobile Division), entitled: "The Propeller Shaft or Hooke's Coupling and the Cardan Joint," which deals with methods of minimising the vibration effects of these reaction couples in motor vehicles.

REVIEWS

STANDARD METHOD FOR THE ESTIMATION OF DIRECT OPERATING COSTS OF AIRCRAFT. Prepared and Issued by the Society of British Aircraft Constructors, London. 1949.

More nonsense has been written about air transport than Lewis Carroll ever wrote. The only comforting thought is that most of what has been written will not live as long as the normal life of a passenger machine.

The Society of British Aircraft Constructors has recognised that nonsense may cease to be funny if it does not lead to an ever-increasing number of air passengers, and they have taken that first serious step of trying to show the potential operator what it will cost him to operate the aircraft the Industry produces.

Let us say at once this is a first-class job, but wisely enough the Society looks upon it as being a prototype "Issue No. 1" for, to quote, "The Society recognises that, with the present rapid development in airline operation, frequent revision will need to be made if the Method is to retain its value. This applies with particular emphasis to the introduction of turbo-propeller and turbo-jet aircraft."

The S.B.A.C. lay all their cards on the table and invite anyone if they have better cards to produce them for the benefit of air transport at large. A careful study of Section 4 which gives a worked example, together with Annex I giving the Calculation of Fuel Carried, Fuel Consumption and Stage Speed, and Annex II giving Computation of Payload, shows that every change appears to have been thought of except that of a change of government.

From the estimation of annual costs to fixed hourly costs, and costs which vary with stage distance or operating conditions, the operator should be able to make a shrewd summary of his chances of succeeding in making a fortune out of air transport.

This publication of the S.B.A.C. inspires confidence by the very thoroughness with which the whole field of operation depending upon the aircraft's characteristics, is covered. The Society is to be congratulated upon taking such a bold step to remove Air Transport Operating Costs from the field of fantasy, which made Alice and the Red Queen go faster and faster without getting anywhere, to the aerodrome of activity from which one really will get somewhere.

One must congratulate the printers on a first class job, too.

Only one small criticism. Must we have in 1.41, ". . . . all the costs of flying, *crewing*, maintaining . . . ." The verb "to crow" but surely not "to crew."—J.L.P.

DYNAMICS OF REAL FLUIDS. E. G. Richardson. Edward Arnold and Co., London. 1950. 144 pp. 96 diagrams. Index. 21s. net.

Discussions between workers in two widely separated fields of fluid mechanics such as, for example, aeronautics and rheology are liable to end in confusion because the methods of approach may be quite distinct. However, it is not really surprising that concentration on such different problems as, say, the determination of the centre of pressure of an aeroplane wing and of the spreading qualities of margarine should give rise to marked differences of outlook. Dr. Richardson's new book is welcome because it covers the whole field of fluid mechanics between these two extremes. His account is unavoidably superficial in parts, because of the short space devoted to each aspect, but numerous references to original papers are given.

An introduction to classical hydrodynamics is given in the first chapter and accounts of boundary layers, periodic flow, turbulence and heat transfer complete the first half of the book. Some of these sections have not taken developments since 1940 sufficiently into account. For example, the mixture length theories of turbulent shear flow are described without reference to the criticisms which have been

advanced against them (but no one has yet proposed a satisfactory theory to replace them). Again, the short section on heat transfer fails to emphasise the importance of Pohlhausen's solutions for the heat transfer from a flat plate along the stream, originally published in 1921, and to recognise that King's theory of convection of heat from a circular cylinder is no longer acceptable.

The second half of the book is devoted mainly to complex applications of fluid mechanics in which experiment is more important than theory. Free surface problems such as cavitation and drop formation are described and a section is given to the behaviour of imperfect fluids, such as elastic liquids and liquids containing solids in suspension. This section contains much interesting material and the author has himself made valuable original contributions in these fields, where the complexity of the phenomena hampers the development of reliable theories.

The book is pleasantly written and gives much information in its one hundred and forty pages. It provides a useful survey of a very extensive field.—H.B.S.

HELICOPTER ENGINEERING. Raymond A. Young. The Ronald Press Co., New York. 1949. $10 net. 285 pp. Illustrated. Appendix. Index.

It has been said that the work involved in the design of a helicopter is 95 per cent. mechanical engineering and only 5 per cent. aerodynamics. The author of this book, however, devotes the first three-quarters of his book to aerodynamics and only the final three chapters to rotor blade stressing and other design problems so that this book becomes the first text book in English on helicopter aerodynamics and its appearance, as such, is a long-awaited event.

Mr. Young has presented a digest of aerodynamic theory previously scattered in many reports, omitting most of the proofs of the formulæ he quotes and giving much of the information in graphical form. This method of presenting essential aerodynamics for helicopter engineers has much to commend it but it places a great responsibility on the author to see that the formulæ quoted are given accurately, that the meaning of the symbols is quite clear and that the assumptions underlying the equations and curves are also given. Unfortunately this book contains so many inaccuracies (more than twenty in the first three chapters alone) that it cannot be used with confidence by anyone not familiar with the subject.

The difficulties in preparing a precis of the theory of the helicopter are formidable. The theory was developed by several authors in different countries using widely differing notations. Much of the early thought was devoted to the gyroplane with the air flow always up through the rotor. Now that the flow is more often *down* through the rotor the signs in many of the equations must be changed or, at least, the positive sense of velocities and angles clearly indicated to the user. Mr. Young does not do this. For example, λ, the inflow ratio, is given in the list of symbols as $\frac{V \sin \alpha - V}{\Omega R}$ (the second V should be V_i) so that λ is apparently positive when the flow is upward through the disc. Later, in equation 23, λ has been taken positive for a downward flow through the disc. If the reader turns for enlightenment to the "definitions" in Appendix 2 he will find a wrong definition: "Inflow ratio—the ratio of the forward velocity of the helicopter and the rotor tip speed." The task of defining clearly the symbols used has not been taken seriously enough and, too often, a formula or diagram copied from a report without "translation." The important reduction of induced flow through the disc in forward flight is barely discussed and the reader is given a power equation (160) which assumes that the induced power is the same at all forward speeds.

The book is well printed and profusely illustrated. (The 163 figures include Fig. 40—Velocity f.p.s. against Velocity m.p.h. It is with relief that we note that a straight line relationship has been obtained!) The appendices include a list of references for further reading.

It is to be hoped that a thorough revision of the book will be made so that it may then be recommended to students of helicopter aerodynamics and to those engineers who are waiting for a survey of the subject.—A.H.Y.

CORRESPONDENCE

AIRCRAFT DESIGN ANALYSIS

There appears to be an error in the paper Aircraft Design Analysis by I. H. Driggs (February 1950 JOURNAL) in the Turbine Engine Power Generator (Section 4) equation (18).

In Section 4, equation (17)

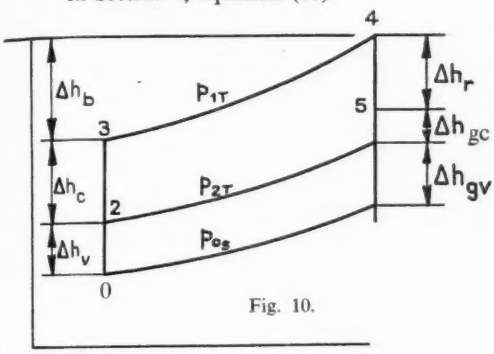
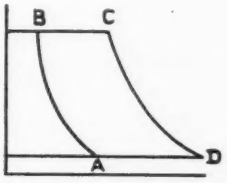


Fig. 10.

$$\Delta h_{gc} = \Delta h_c \left[\frac{\Delta h_b}{h_2 + \Delta h_c} \right] \quad . \quad . \quad (17)$$

This equation is quite easily proved using the air standard efficiency for the ideal gas turbine cycle

$$\eta = 1 - \frac{T_A}{T_B}$$



Mr. Driggs then states: "Similarly, the ideal generator useful enthalpy from the velocity compression cycle can be written as"—

$$\Delta h_{gv} = \Delta h_v \left[\frac{\Delta h_b - \Delta h_{gc}}{h_0 + \Delta h_v} \right] \quad . \quad . \quad . \quad . \quad . \quad (18)$$

Now equation (17) can be generalised as

$$\text{Shaft work obtainable} = \frac{\text{(compressor enthalpy rise)} \times \text{(heat supplied)}}{\text{(enthalpy at compressor entry)} + \text{(compressor enthalpy rise)}}$$

For the velocity compression cycle

"Compressor enthalpy rise" $= \Delta h_v$;

Heat supplied $= \Delta h_b - \Delta h_{gc}$;

Enthalpy at "compressor" entry $= h_0$

And hence (according to Mr. Driggs) substituting in the generalised equation

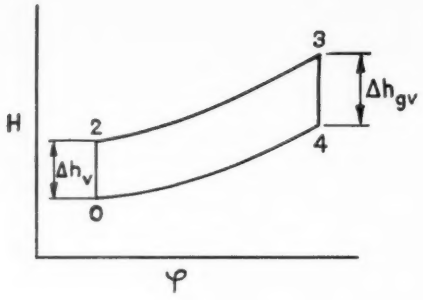
$$\Delta h_{gv} = \Delta h_v \times \frac{(\Delta h_b - \Delta h_{gc})}{h_0 + \Delta h_v}$$

But Mr. Driggs forgets that the velocity compression cycle is not a gas turbine cycle and that the "compressor work" does not have to be supplied by a turbine. Consequently the available enthalpy at exit

$$= \Delta h_{gv} = \Delta h_v \times \frac{h_0 + \Delta h_v}{(\Delta h_b - \Delta h_{gc})} + \Delta h_c$$

To illustrate by an example

Consider an aircraft flying at 600 ft./sec. Let entry conditions be 14.7 p.s.i. and 15°C. Let upper temperature of velocity compression cycle be 600°C. Find enthalpy available for expansion in nozzle due to velocity compression cycle.



$$\Delta T_v = \frac{V^2}{2gJ\varepsilon_p} = \frac{600^2}{2g \times 1400 \times 0.24}$$
$$= 16.75°\text{C}.$$

$$\Delta h_v = C_p \times \Delta T_v \qquad T_2 = 288 + 16.75$$
$$= 4.02 \text{ C.H.U.} \qquad = 304.75°\text{C. abs.}$$

Heat supplied $= H_3 - H_2$ ($= \Delta h_b - \Delta h_{gc}$ in equation (18))
$$= 0.24 (T_3 - T_2) = 0.24 (873 - 304.75) = 136.5 \text{ C.H.U.}$$

$$\frac{T_2}{T_1} = \frac{304.75}{288} = 1.058 = \frac{T_3}{T_4} \text{ since } \frac{P_2}{P_1} = \frac{P_3}{P_4}.$$

Therefore $T_4 = \dfrac{T_3}{1.058} = \dfrac{873}{1.058} = 826$

Therefore $\Delta h_{gv} = C_p (T_3 - T_4) = 0.24 (873 - 825)$
$$= 11.5 \text{ C.H.U.}$$

This is the correct value, and could have been obtained from the corrected formula (18)

$$\Delta h_{gv} = \Delta h_v \left\{ \frac{(\Delta h_b - \Delta h_{gc})}{\Delta h_v + h_0} + 1 \right\}.$$

Using Mr. Driggs' equation $\Delta h_{gv} = \Delta h_v \dfrac{(\Delta h_b - \Delta h_{gc})}{\Delta h_v + h_0}$

we obtain the value $\Delta h_{gv} = 7.50$, a 35 per cent. error.

A. H. Stenning, *Student R.Ae.S.*

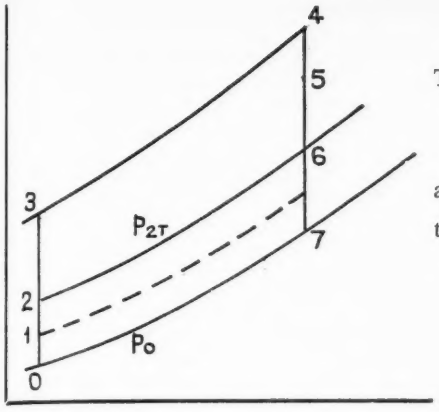
MR. DRIGGS' REPLY

Mr. Stenning is correct in stating that there is an error in equation (18). Evidently the draftsman who filled in the equations after typing made a slip and we did not catch it in proof reading.

The equation should read:

$$\Delta h_{gv} = \Delta h_v \left[\frac{\Delta h_b - \Delta h_{gc}}{h_0 + \Delta h_v} + 1 \right] \quad . \quad . \quad . \quad . \quad . \quad . \quad (18)$$

As Mr. Stenning states the velocity cycle does not depend upon a compressor but his derivation is incorrect since $\Delta h_b - \Delta h_{gc}$ is *not* the heat added in the velocity cycle. Repeating Fig. 10, the derivation of equation (18) follows:



$$\frac{h_2}{h_0} = \frac{h_6}{h_7}.$$

Thus since

$$h_2 = h_0 + \Delta h_v$$

$$h_7 = h_6 - \Delta h_{gv}$$

$$h_6 = h_0 + \Delta h_v + \Delta h_c + \Delta h_b - \Delta h_t - \Delta h_{gv}$$

and since $\Delta h_c = \Delta h_t$,

then

$$\Delta h_{gv} = \frac{\Delta h_v}{h_0 + \Delta h_v}\,[h_0 + \Delta h_v + \Delta h_b - \Delta h_{gc}]$$

$$= \Delta h_v \left[\frac{h_b - \Delta h_{gc}}{h_0 + \Delta h_v} + 1\right]$$

Q.E.D.

I. H. Driggs, *Fellow.*

(Mr. Driggs states that the curves relating to equation (18) *in the paper are correct as they were computed from the correct equation, but he draws attention to an error of like character in Section* 3, *point* 3, *page* 71 (*p.* 17 *in the Advance proof of the paper*) *where the ratio of efficiencies should read* η_{b0}/η_b. *There is a more serious error in the calculation for Fig.* 14. *Although the equations are correct, the calculator misinterpreted them and thus an error was made in plotting the figure. The correct Fig.* 14 *is as shown below. Copies of this correct Fig.* 14 *for those who had preprints of Mr. Driggs's paper may be obtained from the Society.*—ED.)

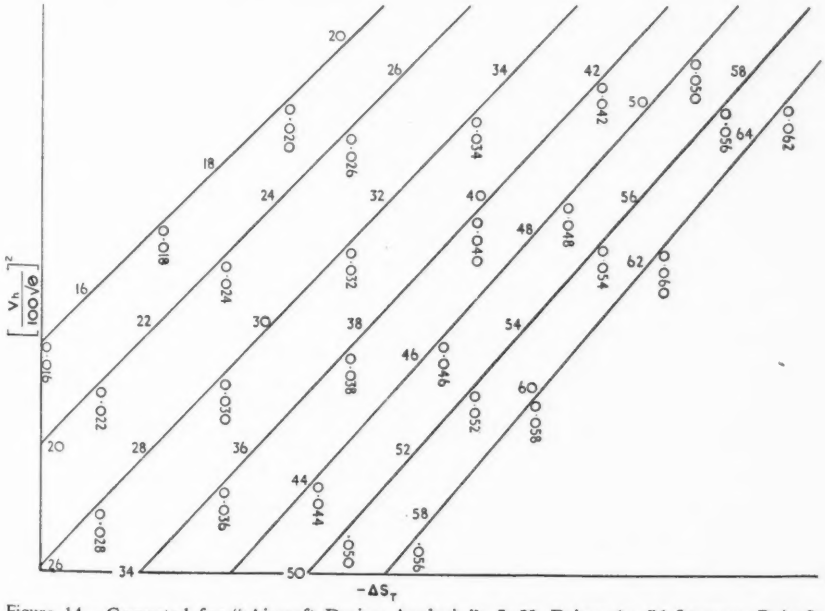


Figure 14.—Corrected for "Aircraft Design Analysis." I. H. Driggs (p. 76 JOURNAL R.Ae.S., February 1950).

COEFFICIENTS OF FRICTION

Miss Pike's valuable summary of information on coefficients of friction of tyres (December 1949 JOURNAL) calls for a few comments, particularly in view of the interest being shown at present in automatic braking systems. These are presumably designed to avoid locking of the wheels, and hence the behaviour of the tyre before locking is of special interest.

Referring to Fig. 2 of Miss Pike's paper, it is important to understand the physical significance of "slip" during braking, which has been investigated by several authors[1, 2, 3]. At relatively low brake forces "slip" is an elastic phenomenon, due to stretching of the tyre wall under the action of brake drag; this corresponds to the portion of the curve near the origin; under these conditions the B.F.C. speed relationship is practically independent of speed and coefficient of friction. As brake force increases sliding begins at the rear of the contact area of the tyre, the extent of the sliding region increasing with increasing brake torque, until eventually sliding occurs over the whole contact area. The transition between rolling and sliding is thus gradual, and there is no justification for the sharp cusp at the point "A" in Fig. 2—a perfectly smooth curve is given by all other authors (*see* particularly Reference 4).

Thus there appears to be no justification for labelling "A" as the "point of impending skidding," which may well be farther along the curve. In any event the important distinction is not between "skidding" and "non-skidding" (the latter being confined to extremely low torques), but between partial skidding and full skidding, since the latter corresponds to total loss of directional stability. Nor does there appear to be justification for saying that tyre wear will be seriously affected by the value chosen for the brake torque coefficient, since partial skidding will be present in any event.

Probably the most important point, however, is the question of the significance to be attached to friction-speed curves. Nearly all the data presented are applicable to the 100 per cent. slip condition, *i.e.* wheels locked. If brakes are designed or used in such a manner as not to cause locking, then it is the friction coefficient at low slips that is the important factor, and this seems to vary very little with speed, as shown for example by the upper curve of Fig. 3 in Miss Pike's paper.

Miss Pike's list of references could be supplemented by Reference 4 below, which appears to be one of the best studies of the subject of friction. There is also a comparatively early German book on the subject, Reference 5. Several comparatively recent Italian investigations are listed in Reference 6.

There is actually a considerable amount of information on rolling resistance in German research reports—*see* References 7, 8, 9 and 10.

1. SCHUSTER and WEICHSLER. Der Kraftschluss zwischen Reifen und Fahrbahn. *Automobiltechnische Zeitschrift*, October 1935.
2. JULIEN, M. L'envirage et la tenue de Route. *Journal de la Société des Ingénieurs de l'Automobile*, April 1937.
3. FROMM, H. Untersuchungen über Seitenkräfte und Wendemomente und ihre Bedeutung für die Verbesserung der Lastannahmen. *Lilienthal Gesellschaft Report 169* (M.O.S., TIB Library Translation 283).
4. FÖRSTER, B. Versuche zur Feststellung des Haftvermögens von Personenwagen Bereifungen. *Deutsche Kraftfahrtforschung, Zwischenbericht 22.*
5. SCHENK. Fahrbahnreibung und Schlüpfrigkeit der Strassen im Kraftwagen Verkehr. *Krayn*, 1930.
6. ARIANO, R. Betrachtungen über die Mechanik des Luftreifens. *Kautschuk* (1941), p. 143.
7. POTTHOFF, H. Der Rollwiderstand von Kraftfahrzeugen. D.K.F., Z.B.11, 1937.
8. KAMM, W. Gesamtfahrwiderstandsgleichung für die Fahrwiderstände von Personenkraftfahrzeugen. D.K.F., Z.B.24, 1938.
9. KLUGE, H., and HAAS, E. Rollwiderstand von Luftreifen. D.K.F. Heft 44, 1940.
10. FRANK and KRANZ. Rollwiderstand von Flugzeugfahrwerken. *Forschungsbericht 1037*, 1939.

R. Hadekel, *Associate Fellow.*

Miss Pike's Reply

I am grateful to Mr. Hadekel for his interesting comments and for a valuable list of additional references.

The paper was written with the emphasis on the friction coefficient under full skidding conditions, since with manual application of the brakes a certain percentage slip cannot be guaranteed, particularly on all the wheels at the same time. If automatic brakes which will prevent locking can be used, interest will naturally centre on the behaviour of the coefficient before locking.

The sharp peak shown in Fig. 2 is representative of the tests of Reference 1 and led to the conclusion that "The evidence appears to indicate the existence of a sharply defined critical point at about 12 per cent. slip." It appears, however, that the peak flattens out with increase of tyre pressure. I agree that there is no real justification for calling this peak the point of "impending skidding," and "critical point" or "peak value" would have been better.

Mr. Hadekel has concluded from Fig. 3 of my paper that the "impending skidding" coefficient is practically invariant with speed. This is not necessarily the case, and other curves from Reference 2 show that it may vary by 0.2 or more over the speed range shown in Fig. 3.

In Figs 18 and 19 of my paper, the values given for tyre pressures have been converted from the German into absolute values, whereas elsewhere in the paper the pressures are values above the atmospheric pressure. This is an error for which I apologise.

E. C. Pike, *Associate Fellow.*

AIRCRAFT DEFECT REPORT PROCEDURE

The following comments are offered on Aircraft Defect Report Procedure (A. H. Greenwood, November 1949 Journal).

Civil procedure in Great Britain is usually much more direct than military, due to several factors. Military maintenance is done at a large number of widely dispersed stations, whereas civil maintenance is largely done at centralised bases, which considerably facilitates the compilation of defect records. Also, the central organisation collecting, analysing and initiating action on the defects is part of the same organisation as the operators, and responsible to the same management; whereas in the military sphere the central organisation (RDA Defects, Ministry of Supply) has no direct connection with the operators.

Service procedures are not usually as elastic as their civil counterparts, and the more cumbersome procedure inevitably leads to delay in rectifying the defects, with the result that engineer officers often do not raise official defect reports for other than really serious defects. Units have been known to refrain from reporting defects because of a suspicion that their efficiency was judged by the number (or rather, lack of) Forms 1022 raised, and have had to be advised to raise one in order to obtain sanction for remedial action.

Such factors as these go far to explain why, on a civil version of a military type, many defects occur on components which are stated to have given no trouble in service use, although another important factor here is the much higher utilisation obtained from civil types which renders a defect occurring at, say, 500 hours very much more serious on the civil version.

The firm's service representatives can do much to alleviate this; they must work in close conjunction with the Service personnel and know which defects are, and which are not, being officially reported, so that they can report them or elaborate on them if necessary. They must be free to give their own uncensored version of the cause of the trouble. Service representatives cannot be maintained at every unit throughout the service life of the aircraft, so inevitably much information that would be of use is lost to the designing firm.

My comments on the four points suggested as the objects for a review are:—

(1) While appreciating that the user has a responsibility to supply information, the maker must press for the information in the form in which he wants it. To do this efficiently necessitates an engineer on the design staff whose responsibility it is to collect such information, interpret the lessons to be drawn from it, and pass these on to the designers. This engineer should also have the responsibility for ensuring that past defects are not repeated in current designs.

(2) This is extremely important from the morale standpoint, as if no reply is forthcoming on past defects, future defects will not be reported.

(3) It is doubtful whether the intermediaries can be made to move any faster; preferably they should be eliminated or by-passed, particularly if, as in many cases, they are only post offices.

(4) Much of the delay is due to remedial action on defects having to take its turn with modifications for operational reasons, improved performance, and so on. It is practically impossible, if the normal procedure is adopted, to embody a modification in under six months from the date of reporting the defect.

All this leads to the obvious conclusion that the ideal is to eliminate all defects in the design stage. While this is not practically attainable, much more could be done than is at present if past lessons were taken to heart, and this will only be done in earnest if there is some form of penalty, such as the free replacement of parts or rectification of defects, in the contract. Something much more watertight than the present form of guarantee is required.

E. R. Major, *Associate Fellow.*

Mr. Greenwood's Reply

I am grateful for the amplification which Mr. Major's letter gives to several points in the paper. Civil procedure in this country is fairly direct, but it must be remembered that the average aircraft manufacturer sells his aircraft abroad, and is vitally interested in the effect of climatic extremes. It is still difficult to get reliable information from overseas, unless a firm's representative is stationed at the operator's headquarters.

The implication of comment (4) is, of course, that the number of modifications must be reduced. It is possible that the operator can assist materially in this reduction by an early and firm formulation of design requirements, and by adopting a degree of operational and performance standardisation.

A. H. C. Greenwood, *Associate.*

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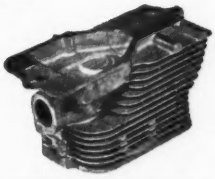

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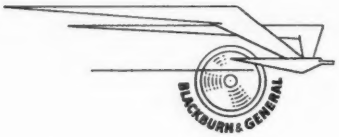

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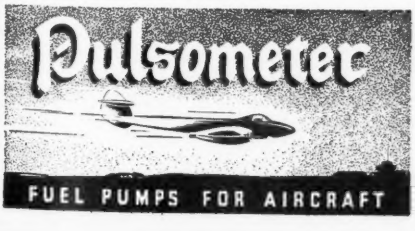

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